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AEROPLANE
RUCTURAL DESIGN

T. H. JONES & J. D. FRIER

AEROPLANE STRUCTURAL DESIGN

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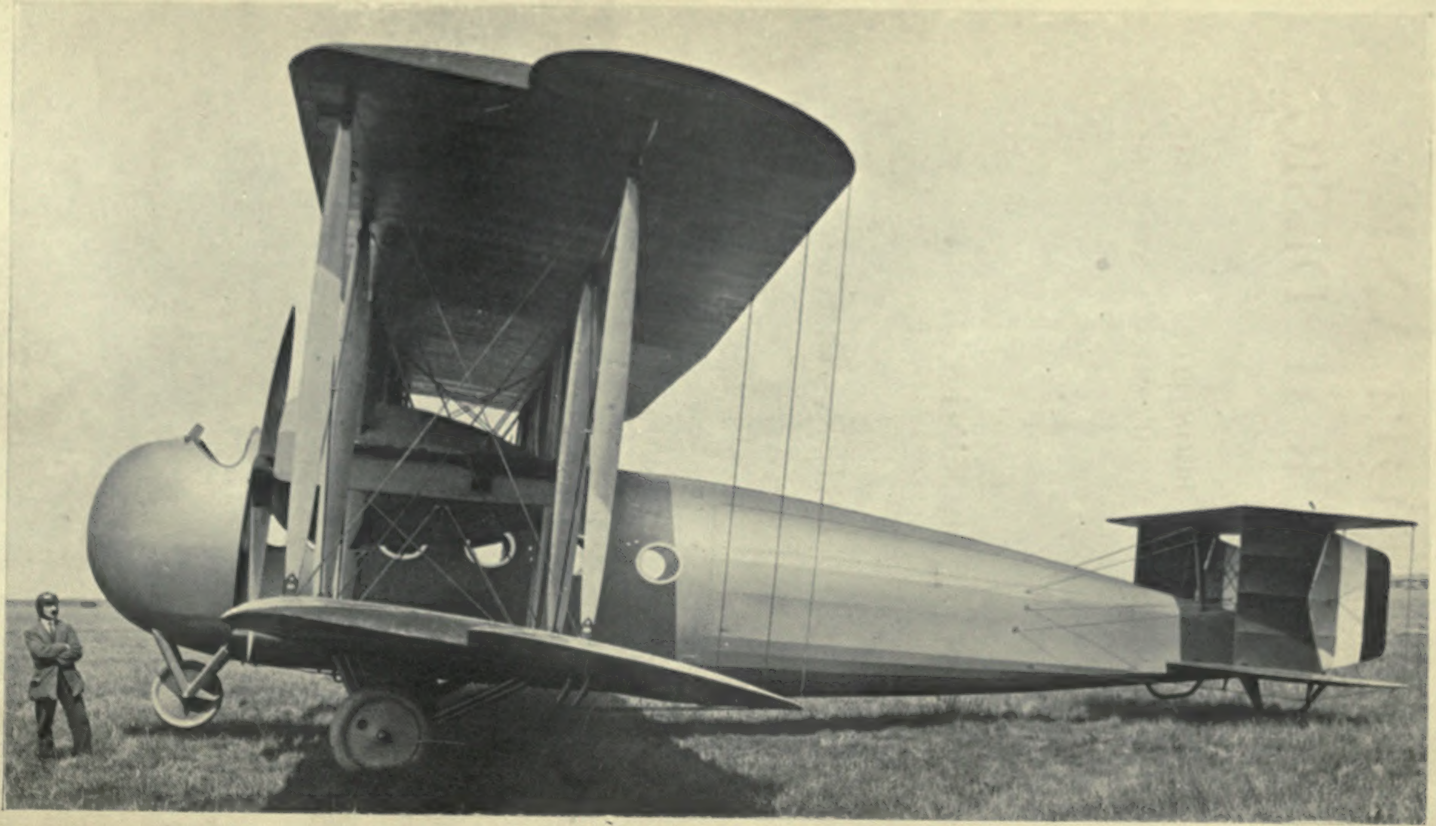

PLATE 1 VICKERS' "VIMY" (COMMERCIAL) TWO 375 H.P. ROLLS-ROYCE ENGINES

AEROPLANE STRUCTURAL DESIGN

A BOOK FOR
DESIGNERS, DRAUGHTSMEN & STUDENTS

BY

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PREFACE

Several valuable treatises are already published dealing with the problems of sustentation, stability, and aerodynamics generally, but it will readily be admitted that if the safety of heavier than air craft is to be assured, it is essential that the structural strength of the aeroplane should receive adequate attention.

Much of the existing published matter dealing with the strength of construction of aeroplanes is thought to be inadequate from the standpoint of the designer and stress draughtsman, and it is the Authors' experience that draughtsmen generally have more or less hazy ideas and views regarding the loading of aeroplanes, and more generally the methods of estimation of the strength of aeroplanes under such loading. It is, therefore, with the idea of dispelling this gloom that the proposal to publish the following matter in as palatable a form as possible, originated. To the purely academical and mathematically inclined reader, much of the matter may be lacking in direct interest, but it is hoped that the designer or the draughtsman who is responsible for the strength calculations of aeroplanes from a practical design point of view may derive considerable benefit. Utility, simplicity, and clearness throughout have been consistently aimed at, and it is hoped that this end also has been attained.

Much matter such as is usually found in the standard textbooks on Structures, Graphics, and Strength of Materials, has been purposely omitted from this volume, it being thought that useless duplication of such matter would be less acceptable from the reader's standpoint.

No one book in any science can ever be said to have reached finality, and the Authors realize that in such a science as Aeronautics, which is even yet in some respects only in its infancy, many advances are bound to be effected in the light of further experience.

We desire to put on record our thanks to the following—

Aircraft Manufacturing Co.
Austin Motor Co.
Bristol Aeroplane Co.

Director of Research, Air Ministry.
Handley-Page, Ltd.
McGruer Spar Co.
Superintendent, Royal Aircraft Establishment.
Supermarine Aviation Works, Ltd.
Vickers, Ltd.

All of whom have been unstinting in their offers of valuable data, photographs, blocks, and descriptive matter, which they have generously placed at our disposal for the purpose of this publication.

We also wish to express our thanks to Prof. Sir Ernest Petavel, K.B.E., D.Sc., F.R.S.; our friends H. A. Webb, M.A., A.F.R.Ae.S., and J. Kenworthy, B.Sc., A.F.R.Ae.S., for timely suggestions, which have been distinctly helpful.

It is too much to hope that this edition will be entirely free from errors, and any intimation of these, together with further suggestions, will be much appreciated.

J. D. F.
T. H. J.

BIRMINGHAM,
8th May, 1920.

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AEROPLANE STRUCTURAL DESIGN

CHAPTER I

INTRODUCTION

Although it may appear to be unnecessary, yet it is thought advisable to begin by glancing at the several components of an aeroplane, casually noting their functions, and describing very briefly any points of interest regarding them.

This is followed by a typical example of an aeroplane of medium size in which all the strength calculations necessary to the complete design of the aeroplane are clearly shown, as this is more instructive than a bare statement of methods without examples showing their application.

The factors of safety or loads used, are based on the results of model experiments, full scale work, or mathematical investigation by various authorities.

For a given wing area, it is thought that a commercial aeroplane should be just as strong in its structure as a war machine. The latter must be able to effect a safe landing when various wires or members have been damaged through enemy action; while in the case of the former, it is important that the life of the machine should be as long as possible on grounds of efficiency and economy.

MAIN PLANES OR WINGS

For all practical purposes, it is sufficient to say that the wings, or planes, are the only supporting surfaces deserving notice, and, in fact, it is usual to neglect other possible sources of lift altogether. The wings depend for their lifting power upon the peculiarity of their chordal section, which possesses the unique property, when set at a few degrees of incidence, of giving a large ratio of lift to drag.

Different aerofoil sections possess this property in varying degrees, but for any one section, the maximum value of lift to drag is generally obtained for an angle of incidence somewhere between 3 and 5 degrees. In the aerofoil section known as R.A.F. 15, it has been found by experiment that the maximum ratio of lift to drag is obtained when the angle of incidence is in the vicinity of 3 degrees, when the ratio has a value of 17·3.

The magnitude of the lifting force depends, among other things, upon the relative speed of the machine, and the incident air meeting the planes. It follows then that there is a definite minimum speed of the machine relative to the air, which must be attained in order that it may leave the ground.

This landing, or "stalling" speed varies, however, with different machines, being dependent upon the lift coefficient of the aerofoil and the wing area. It is a significant fact that the greater proportion of the lifting force is obtained from the upper surface of the aerofoil. Here, there is a negative pressure or suction, which is large compared with the positive pressure at the under surface of the aerofoil.*

The construction of the plane takes the form of two main spruce spars connected by a series of compression ribs and lighter ribs of spruce, having the shape of the aerofoil section. Round cross bracing wires are inserted between the front and rear spars in the several bays formed by the compression struts, the wires being attached to metal fittings bolted securely to the spar. The spars may be of solid spindled section, or of built-up hollow section. Occasionally, spars, built up of metal strips rolled to various shapes of section, are used, and in a few cases complete metal planes have been used with great success.

TYPES OF FUSELAGE

The Girder Type

So called because it is built up in such a form that the sides of the structure resemble an ordinary lattice or braced girder.

It is composed of longerons, cross struts, and wire bracing, the number of longerons being dependent upon the cross-sectional shape of the fuselage (*i.e.*, triangular shape has three longerons, rectangular shape has four longerons, and so on). The general shape, however, is rectangular.

---

* For further information the reader is referred to *The Properties of Aerofoils* (Judge), published by Messrs. Pitman & Sons.

The connection between the longerons and cross struts is accomplished by joints of steel plates and bolts, to which the bracing wires are attached.

The longitudinals and cross struts are usually made of spruce or ash, and the bracing of round swaged steel wires. Any necessary adjustment may be made by means of turn-buckles on the bracing wires, for it is found that after a few flights this type of fuselage usually requires "tuning."

It is essential also to insert diagonal or bulkhead bracing to withstand any twisting tendency on the structure.

When well designed and carefully constructed, it forms quite a light, strong, and elastic body for the machine.

This girder type of fuselage is sometimes made throughout of metal tubing with wire bulkhead bracing at intervals, and possesses the advantage over the one just described in as much as no adjustment is ever necessary in the structure when it is once constructed.

A variation of this form may be made by using pressed channel or angle metal instead of tubing. These form quite strong and durable frame structures, and are not much behind in the matter of weight.

These skeleton frames must always be covered in some way (usually by doped linen fabric), in order to minimize the head resistance of the machine.

Monocoque Types

This used to be and still is, with many people, a very favourite method of construction for the aeroplane fuselage. It is usually composed of several light bulkheads made of spruce, which are connected by a number of more or less slender wooden longerons, the whole being covered in by thin ply-wood which plays the part of bracing as well as covering or fairing. In some cases, light metal bulkheads are used, and the covering made of thin but strong metal sheet, stiffened by longitudinal ribs. Whether metal or timber be used, no wire bracing is usually required, since the covering is supposedly strong enough to resist any tendency to deform. There are, however, difficulties to be overcome in these types, since it is not always easy to make attachments for wing spars, engine bearers, tail planes, and the like where considerable local strength is required.

Whatever type of fuselage is adopted, the outer shape should be good aerodynamically so that its head resistance, which is often as much as one-half the total resistance of the whole aeroplane, may be kept within reasonable limits. Projecting parts, gaps in the surface, sudden changes in outline, should all be avoided as much as possible, in order that the shape may approximate to a streamline form. The tractor type of fuselage, by its tapering away in the rear, lends itself more readily to a good aerodynamic form than that of the pusher type, where the body is shorter and blunter ; the connection between wings and tail unit being made by means of outriggers, wires, and struts.

THE PROPELLER

The propeller, tractor, or airscrew provides the necessary means of propulsion by discharging the air backwards through its own disc, thus producing forward motion of the machine. The cross section of the propeller blade is usually an approximate aerofoil section.

Materials used in the construction of propellers are mahogany and walnut built up in laminae, well glued together, and afterwards carefully shaped. Two-bladed and four-bladed airscrews are in general use. Three-bladed ones have been tried, but have not been very successful.

The term " pitch " in connection with airscrews is capable of being defined in several ways. It is suggested that reference be made to "*Airscrews*," by Riach, where these definitions are given. Most propellers are at present of " constant pitch," but a few of " variable pitch " have been designed and used with good results. Here, the blades are rotated about their own axis, so that the angle of attack of the blades may be altered in flight to increase the airscrew efficiency with altitude.

Metal propellers have been attempted, but difficulties of manufacture have prevented developments in that direction.

LANDING GEAR

Before " lifting," the machine will " taxi " along the ground on its landing gear or chassis, which consists generally of two or more tyre-covered wheels on an axle connected to the fuselage, or in some cases to the lower planes, by elastic or steel spring shock absorbers. These shock absorbers, which vary in construction and design, must be strong enough to enable the machine to land without any danger to the fuselage due to the shock of impact.

On one or two types, an oil dashpot arrangement is used to damp out vibration and absorb shock in landing.

CONTROL SURFACES

In order to secure safety in flight, several stabilizing or control surfaces must be provided. These consist of ailerons, fins, tailplane, elevators, and rudder, of which the fins and tailplane are generally fixed surfaces, whereas the ailerons, elevators, and rudder are movable surfaces controlled in flight by the pilot. Very often, however, the tailplane is made "adjustable" so far as angle of incidence is concerned, and sometimes the fins are capable of slight lateral adjustment, but this latter cannot be made during flight.

The position, shape, and setting of any surface depends upon the type of machine and for what purpose it is designed.

For instance, the tailplane section is generally a so-called non-lifting surface (*i.e.*, it is simply a symmetrical section); but in flying-boat work, where the engine is set very high relative to the C.G. of the whole machine, a strong nose-diving moment is introduced, and in order to counteract it, the tailplane section may be an inverted aerofoil section, so that while the engine is running there is always a down load on the tailplane.

Lateral controllability of an aeroplane depends mainly upon the size and shape of the ailerons, although other factors enter into the question to a lesser extent, namely, shape of main planes and moment of inertia of the machine.

The construction of the tailplane and elevators is of the same kind as in the wing and ailerons respectively, and does not call for any further comment.

In the construction of the rudder and fins, metal tubing and spruce ribs are frequently used, and a covering of doped linen fabric applied.

REAR SKIDS

These vary very much in type and construction, and also in the materials used. Very frequently, all-metal skids are used, having steel springs to absorb the shock, whilst other types are constructed entirely of ash, with rubber cord absorbers.

Both types may or may not be controllable from the cockpit ; and, when steerable, the control wires are usually attached to those of the rudder, so that simultaneous movement is effected.

LOAD FACTORS

Load factors must be such that they will control design to such an extent that the aeroplane may withstand the most severe loads which can possibly occur in flight without the slightest sign of failure. For this purpose, the criterion of strength must be the stress which produces the first sign of compressive failure, rather than the breaking stress. This corresponds with the elastic limit. In the following calculations, the load factors are founded partly upon experimental investigation and partly upon reasonable assumption.

The empirical curve in Fig. 2 gives factors which cover all ordinary loads incurred in flying, diving, etc., and it is only in cases of very quick manoeuvre, heavy stunting, as in military work, that the loads will, if ever, exceed those given by the curve.

In any case, so much depends upon the size and type of the machine, and the way in which the pilot operates the controls, as to how much extra load is imposed upon the aeroplane in flight, that no present-day designer would think of designing a machine to withstand ten or twelve times load in order to attempt to make it fool-proof. It may be that some official publication will shortly be made as to the factors recommended to apply to the design of different sizes and types of machine.* The methods of design given in the following work, however, are capable of application under any system of factors which may be adopted, and they have been in use for several years with satisfactory results.

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* This information is now available in the form of a report of the Advisory Committee for Aeronautics, issued by the Air Ministry, a reprint of which appears in the appendix to these pages.

The general conditions of loading which ought to determine the design of a machine are—

(1) Normal horizontal flight with the centre of pressure at ·5 of the chord, which is the common condition of flying.

(2) Flattening out from a vertical nose dive, or coming out of a loop, in which case the centre of pressure may be as far forward as ·3 of the chord. In this case so much depends upon the rate at which the flattening out takes place as to the number of times normal load which will be imposed upon the machine. The speed in a vertical dive may become very high until it reaches a limit depending upon the resistance of the machine. If now the machine be suddenly brought to the stalling angle (*i.e.*, the angle where maximum lift is obtained), the lift will be proportional to the square of the velocity attained in the dive; and the number of times normal load will therefore be proportional to the ratio of the two velocities (*i.e.*, the ratio of velocity attained in the dive), to the stalling speed, if the change from the one to the other is brought about very quickly by the pilot.

Thus, the load factor will be $\left(\frac{V_d}{V_s}\right)^2$ where V_d is the velocity attained in diving, and V_s is the stalling speed. This sudden flattening out is very difficult to effect except in a very small machine, and the careful pilot will avoid it even then. Generally, the maximum speed V_d is arbitrarily fixed, and the corresponding factor found as described above, the machine then being designed to stand up to that factor of loading. For example, the maximum speed of the aeroplane considered herewith is fixed at 115 miles per hour, the stalling speed being 51 miles per hour. Hence the factor required is $\left(\frac{115}{51}\right)^2 = 5$ nearly, which corresponds to the empirical curve, Fig. 2.

(3) Down load due to downward gusts, or the possibility of reversed loading in a loop. This case covers the case of very steep nose-diving, where the centre of pressure is probably well behind the rear spar, thus producing down load on the front truss and up load on the rear.

If any down load occurs on the rear truss it is comparatively small, and it is usual to work out the front truss under a loading of three times normal load with the centre of pressure at ·3 of the chord, and only to roughly estimate the loads on the rear truss or neglect them altogether.

Turning in a horizontal circle may induce excessive loads depending upon the strength of the pilot in making the machine bank steeply. Generally speaking, this case is covered by (2), and is hardly worth further consideration from our standpoint.

Flying in a spiral, and spinning, zooming, etc. These cases do not demand special consideration regarding the manner in which they impose loads on the structure, since it is only in very exceptional cases that the magnitudes of such loads are in excess of those already covered by (1), (2), and (3).

Reference to the Advisory Committee's Reports is suggested where these " stunting " cases are dealt with from an experimental standpoint, and it is for the intelligent designer or draughtsman to sift all the valuable information contained in these publications and extract the portions immediately useful to him.

Application should be made to—

H.M. Stationery Office,
Kingsway,
LONDON.

CHAPTER II

WINGS

It is absolutely essential that the wings be as light as is possible, consistent with strength, and probably it is with this end in view that considerable pains are taken in the estimation of the stresses and loads in the wing structure.

Before we can make any estimation of stresses, we must know in what way the load is imposed on the structure, and the degree of accuracy to which these stresses can be estimated depends largely on an accurate knowledge of the conditions of loading, and the probable maximum intensity of loading on the wings. Our knowledge of the conditions of loading is dependent to a large extent on experiments with small-scale models of aerofoil sections, as well as full-scale flying experiments. From the results of these experiments it is found—

(1) That it is possible for the centre of pressure to vary between about ·3 and ·5 of the chord.

(2) That the intensity of loading on the wing tip is generally less than at other parts of the wing surface.

(3) That it is possible under some conditions of flight to have an intensity up to seven times load, or even larger. This, however, varies with the size and type of the machine; for instance, it is well known that fast small machines require a fairly large factor because of the demands made upon them in quick manoeuvreability, which prodrces large accelerations and, consequently, heavier loading. On the other hand, a larger slower machine is not capable of being so quickly manoeuvred, and therefore requires a smaller factor.

Making use of this knowledge, the remainder depends upon the accuracy of the methods employed in the calculations. The importance of accuracy in the estimation of stresses in the wing structure of an aeroplane cannot be overlooked, and it demands considerable attention on that account.

The ordinary methods give results for the spars which are fairly good, excepting when heavy compressive end loads are taken by the spars, when they are more or less unsafe to use generally.

Again, the exact mathematical solution is not easy to apply to the case of design, but is more useful for merely checking design already completed.

Experience in the strength calculations of many aeroplanes has led us to the conclusion that the best way of estimating the size of aeroplane spar sections for a machine of definite weight and for a given factor of safety is—

(1) Find as accurately as possible the loading on the planes and deduce an approximate B.M. diagram. Make an intelligent guess at the spar section, whose approximate moment of inertia will accommodate the maximum bending-moment obtained from the B.M. curve already roughed out, taking account of end load but neglecting E.L.B.M. (End Load Bending-Moments).

(2) Apply a more accurate method by means of which a bending-moment diagram is evolved which includes end load bending-moment. Find more accurately the moment of inertia of the section which it is thought from the approximate consideration will suit the case, and obtain the factors ; taking into account direct end load, also paying attention as to where the spar is solid, and where spindled out to form the ordinary I section which used to be a common section of aeroplane spars.

It may be mentioned here that the modern tendency is to use box spars, having side webs about $\frac{1}{4}$ in. thick glued to flanges of, say, $\frac{3}{4}$ in. thick, the finished spar being carefully wrapped with glued fabric.

The idea of using these spars originated from the shortage of larger pieces of spruce required by the old type of spar, but box spars possess a few advantages which make them popular with present-day designers.

Generally, before any strength calculations are made, the overall sizes of the aeroplane have already been fixed up from aerodynamical and other considerations, that is to say, the span, chord, and gap of the wings, as well as the approximate shape of the plan form of the wing, and the spacing of spars and struts, are known.

As an example of the application of the various methods of estimating the stresses in an aeroplane it is proposed to consider the case of a medium-sized machine which will need the application of all the methods of calculation described and suggested later. The arrangement drawings of this machine are shown in inset at page 8, from which all the essential dimensions can be scaled.

It is hoped that by this means the methods suggested can be most readily followed in their application to design. For aeroplanes of larger or smaller sizes there is no difference in the principles of the methods employed, and there will, therefore, be no difficulty in applying them to all sizes and types of aeroplanes.

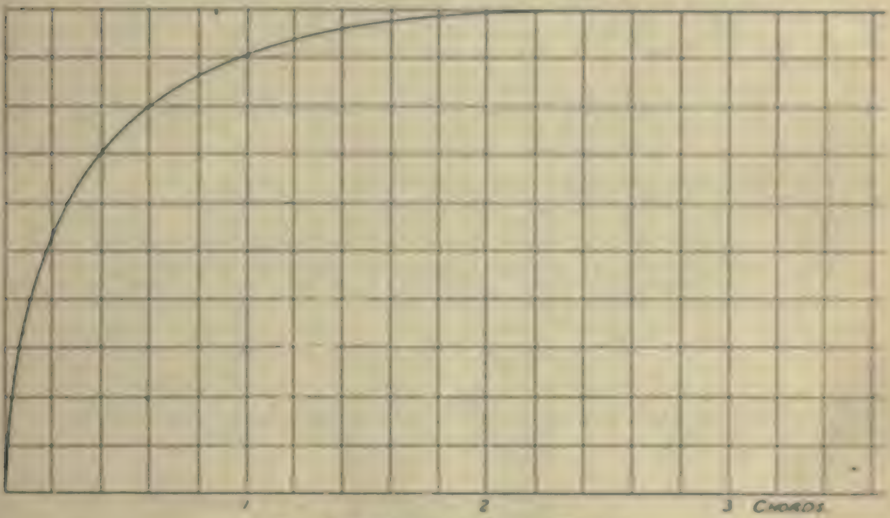

FIG. 1.—GENERAL LOAD CURVE

The machine to be used as an example is a cargo-carrying aeroplane, with the pilot seated immediately behind the cargo. It makes no real difference of course in the strength calculations whether the machine is of a military type or commercial type.

Taking the wing area of the aeroplane under consideration, it is found from Fig. 2 that the factor required is about five times load. This curve need not be accepted as applicable to every class of aeroplane, but it is quite reliable for moderately fast machines, having no special features, otherwise the type must be considered independently of the wing surface. The following particulars should always be tabulated when

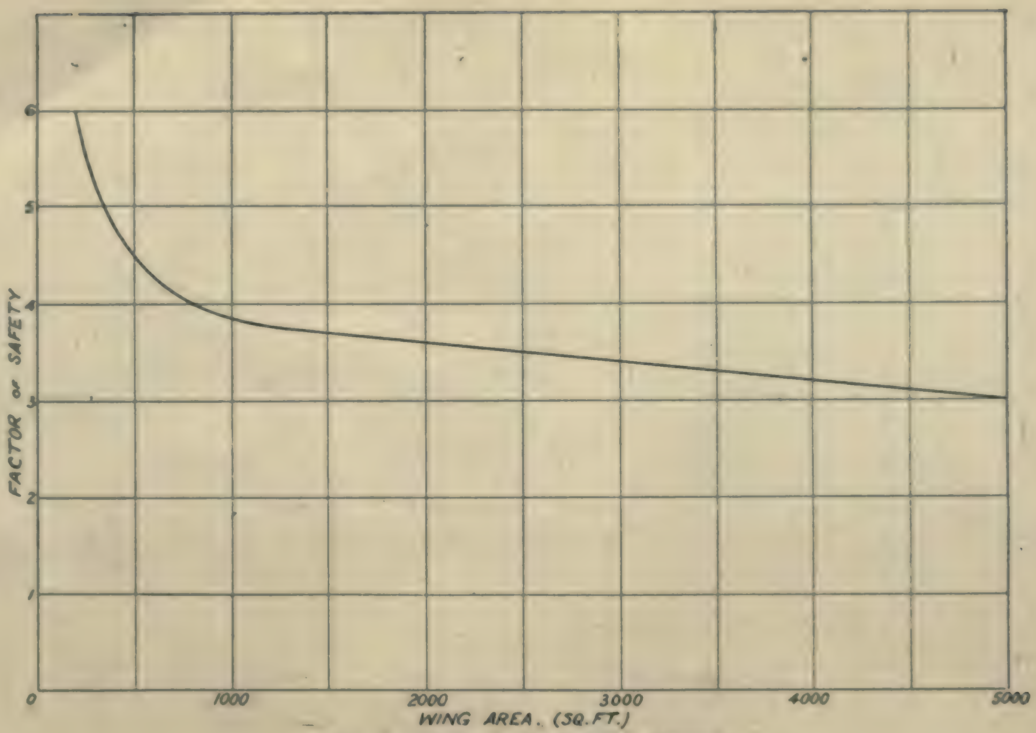

FIG. 2.—FACTOR OF SAFETY CURVE

commencing the calculations, so that reference can always be made to them throughout the work—

Total weight of machine when fully loaded, including cargo, pilot, petrol, etc.	2330 lbs.
Total weight of wings, including struts and wires	330 lbs.
Normal load of machine	2000 lbs.
Area of top plane (with 2 ailerons, each 15 sq. ft.)	172·5 sq. ft.
Area of bottom plane (with 2 ailerons, each 15 sq. ft.)	160·0 sq. ft.
Total wing surface	332·5 sq. ft.

STAGGER OF, AND LOADING ON, PLANES

The majority of modern aeroplanes have positively staggered wings, that is, the top planes are forward of the bottom planes; many others are entirely without stagger; whilst a few are given a negative stagger.

It has been found from tests conducted at the N.P.L. that if the wings are entirely without stagger then the relative efficiency of the bottom planes to the top planes is about 80%, and that if the wings are given positive stagger, this relative efficiency is increased to about 85%.

The stagger of the wings of the machine under consideration is positive; hence the equivalent monoplane surface for the whole machine is—

$$172{\cdot}5 + 160 \times {\cdot}85 = 308{\cdot}5 \text{ sq. ft.}$$

The mean gross load on the top plane is, therefore, equal to—

$$\frac{\text{Total weight of machine}}{\text{Equivalent total surface}} = \frac{2330}{308{\cdot}5} = 7{\cdot}56 \text{ lb. per sq. ft.}$$

and the mean gross load on the bottom plane is—

$${\cdot}85 \times 7{\cdot}56 = 6{\cdot}42 \text{ lb. per sq. ft.}$$

The weight of the wings per square foot—

$$= \frac{330}{332{\cdot}5} = {\cdot}99 \text{ lbs. per sq. ft.}$$

hence the mean net load on the top wing is—

$$7{\cdot}56 - {\cdot}99 = 6{\cdot}57 \text{ lbs. per sq. ft.}$$

and the mean net load on the bottom plane is—

$$6{\cdot}42 - {\cdot}99 = 5{\cdot}43 \text{ lbs. per sq. ft.}$$

The net load per square foot is the one to be used in these calculations, since it must be clear that only the normal load of the machine is able to impose stresses in the wing structure itself (*i.e.*, the lift required to support the wing structure does not impose stresses therein to any extent).

POSITION OF CENTRE OF PRESSURE

Fig. 3 shows the two extreme positions of the centre of pressure relative to the wing spars, for which the two ordinary cases of flying are now considered. It is well worth setting out a full size drawing of the aerofoil section to be used, and inserting thereon the positions of the front and rear spar centre lines. This will enable the particular depth of spar sections to be measured direct from this full size drawing. These depths will be required when the moment of inertia, area, and modulus of the section come to be calculated.

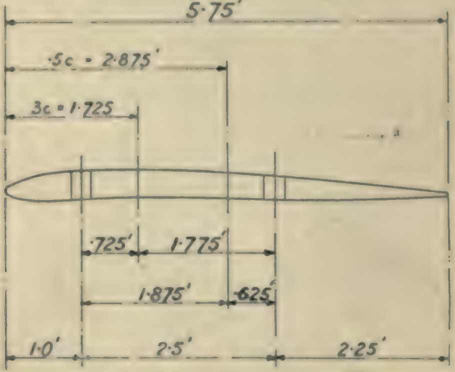

Fig. 3.—Centre of Pressure Diagram. Extreme Positions

The case when the centre of pressure is forward is the one in which the front truss is most heavily stressed; on the other hand, with the C.P. back, the rear truss is most heavily stressed. In both of these cases the top and bottom drag bracing must be considered. There are one or two other special cases of flying for which the wings must be considered, but these will be fully discussed later.

Centre of Pressure Forward. Case 1

When the centre of pressure is forward, then the aeroplane is flying with a large angle of incidence, in the neighbourhood of 16 degrees, and most of the load is taken on the front spars and struts, and in order to find what proportion of the load is taken on the front truss, the load curves for the wings must be examined.

Since the centre of pressure is capable of having positions at ·3 and ·5 of the wing chord, it is *first* proposed to consider separately the two cases of flying—

Case 1.—Centre of pressure forward (*i.e.*, at ·3 of wing chord measured from the leading edge).

Case 2.—Centre of pressure back (*i.e.*, at ·5 of wing chord measured from the leading edge).

Load Curve

Many experiments have been carried out to determine the way in which the air pressure is distributed along the length of the wing; notably these experiments were carried out by the National Physical Laboratory and the R.A.E., where a series of curves showing how the load curve varies for different shapes of wing tip were evolved.

From the results of these experiments a mean curve was adopted by the R.A.E., and regularly used by them.

Fig. 1 is a slight modification of the R.A.E. curve, it is actually a mean between R.A.E. and N.P.L., and is suggested as being quite a suitable load curve which will give spars of the requisite strength. It has been found that the best basis on which to set out the load curve is that of chords, so that the curve will be applicable to every size of chord on every machine; thus the load curve shown is arranged so that 2·5 in. horizontally represents one chord, which enables the linear scale to be determined for any particular chord length.

In the example the chord length is 5·75 ft., hence the scale of the load diagram is—

$$2{\cdot}5 \text{ in. represents } 1 \text{ chord} = 5{\cdot}75 \text{ ft.}$$

$$\therefore 1{\cdot}0 \text{ in. represents } \frac{5{\cdot}75}{2{\cdot}5} = 2{\cdot}3 \text{ ft.}$$

The way in which the intensity of loading falls away toward the tip of the wing is well represented in Fig. 1, and it is not difficult to modify the curve to suit any special case where the chord is variable. Such a case in which this modification is necessary is that in which a portion of the trailing edge is cut away at the centre section for the convenience of the pilot. The particular shape of diagram showing the variation of load intensity along the chord is not of importance, provided the positions of the centre of pressure are known, consequently it is not discussed here. The shape of such a curve is, approximately, a rounded triangle.

LOAD DIAGRAM FOR TOP WING

The length of diagram corresponding to the distance from the extreme wing tip to the centre line of the machine can now be set out, indicating also the positions of the interplane struts; the extreme tip of the wing coinciding with the end of the diagram. (*See* Fig. 5.)

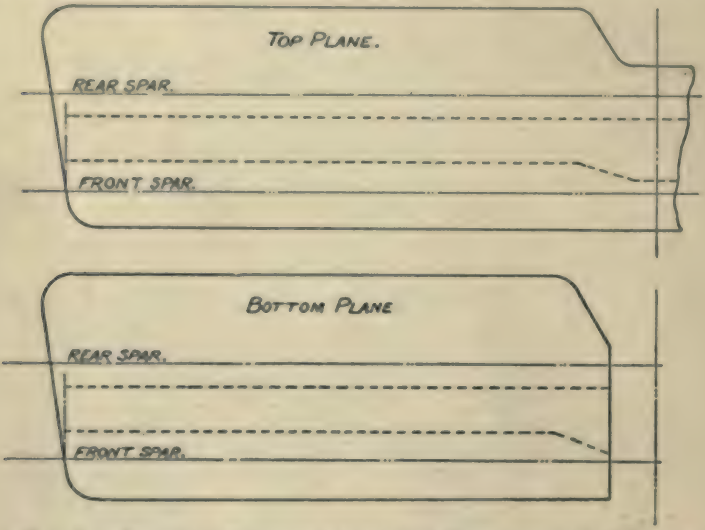

FIG. 4.—DIAGRAM SHOWING ASSUMED MOVEMENT OF CENTRE OF PRESSURE FROM ·3 TO ·5 OF THE CHORD

Dealing with the top wing first: the load curve diagram must now be modified to suit the shape of wing at the centre section (*i.e.*, the ordinates of the curve are reduced in the proportion of the chord over that portion of wing).

In Fig. 4 is shown a plan of the wings and the position of the centre of pressure at all points of each wing, which will

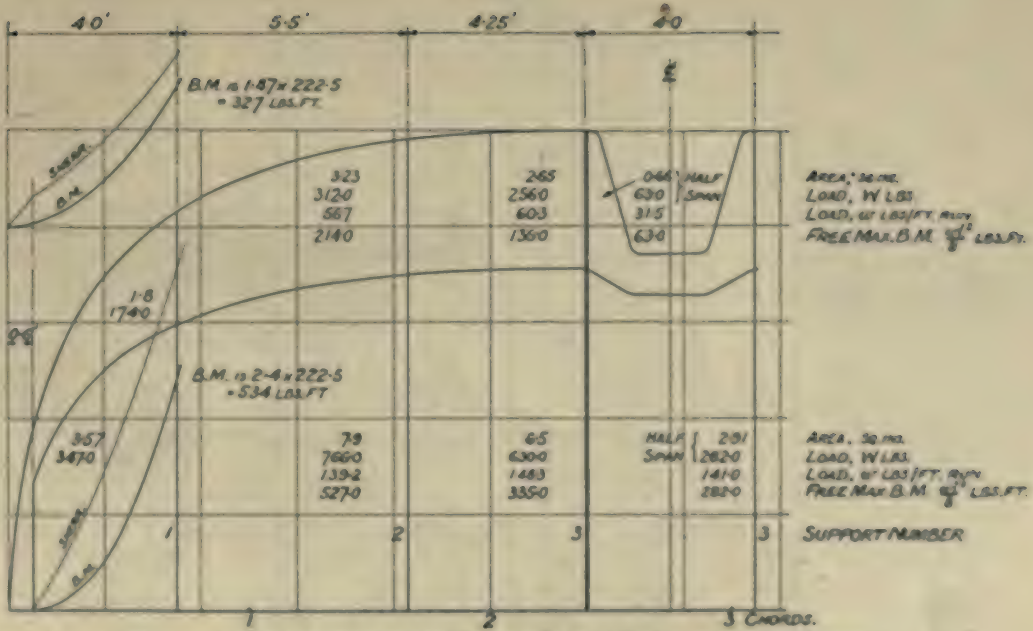

FIG. 5.—LOAD CURVE FOR TOP FRONT SPAR, C.P. AT ·3 OF THE CHORD
5 TIMES LOAD

be useful in setting out the modified load curve for the front spar.

This having been done, the area of the load curve enclosed by this modified outline represents to some scale, not yet known, the total load on one-half of the whole wing, the other half being exactly similar because of symmetry. The proportion of load taken by the front spar must now be shown on this load curve, and if the wing were of rectangular plan form throughout, then the load on the top front spar would be a definite proportion of the total load on the wing *i.e.*—

$$\frac{1{\cdot}775}{2{\cdot}5} = {\cdot}71 \quad (\textit{See}\ \text{Fig. 3.})$$

It should be noticed that for the centre portion the centre of pressure is still assumed to be at ·3 of the chord, although the chord is reduced there, thus, for that portion of the wing, the proportion of load on the front spar is greater than in the case at the outer portion, because the centre of pressure is nearer to the leading edge. The wing tip claims some attention inasmuch as it extends about ·6 ft. beyond the end of the front spar, consequently, any load occurring on that extended portion may be assumed to be taken by the rear spar, and the load diagram for the front spar must therefore terminate at a point ·6 ft. from the end of the complete diagram. (Fig. 5.)

This diagram has now been divided into two portions, the upper portion representing the load on the rear spar whilst the lower portion represents the load on the front spar, for which the scales are now to be investigated.

LOAD SCALES, Top Front Spar

From previous work the mean net load per square foot on the top plane is 6·57 lbs. per sq. ft., and the area of one half the top plane is $\frac{172{\cdot}5}{2} = 86{\cdot}25$ sq. ft.

Hence the load on one half plane—

$$= 86{\cdot}25 \times 6{\cdot}57 = 566 \text{ lbs.}$$

The factor decided upon being 5 times load, the load on one half the top plane—

$$= 566 \times 5 = 2830 \text{ lbs.}$$

Now the area of the load diagram, Fig. 5, representing the load on both front and rear spars, is 29·22 sq. ins., hence 1 sq. in. of the load diagram is $\frac{2830}{29 \cdot 22} = 97$ lbs.

The area of the load diagram representing the load on the top front spar = 20·88 sq. ins., then the load taken by the front spar is equal to 20·88 × 97 = 2025 lbs.

The mean height of this diagram is—

$$\frac{\text{Area of diagram}}{\text{Length of diagram}} = \frac{20 \cdot 88}{6 \cdot 6} = 3 \cdot 17 \text{ ins.}$$

The mean load per foot run on the front spar =

$$\frac{\text{Total load on spar in lbs.}}{\text{Length of spar in feet}} = \frac{2025}{15 \cdot 15} = 133 \cdot 5 \text{ lbs. per foot run.}$$

It therefore follows that the load scale is 1 in. =

$$\frac{\text{Mean load per foot}}{\text{Mean height of diagram}} = \frac{133 \cdot 5}{3 \cdot 17} = 42 \cdot 1 \text{ lbs. per ft. run.}$$

The areas of the front spar load diagram, between the outer and inner strut, between the inner strut and the body strut, and in the centre bay, are now estimated either by counting the squares or by means of a planimeter, such areas being inserted on the load curve in the respective bays to which they apply, together with the loads corresponding to them, the scale being 1 sq. in. = 97 lbs.

If the load in any bay be divided by the spacing of the struts in that bay the mean load per foot in that particular bay is obtained. For example: referring to Fig. 5, first bay, between the outer and inner struts the area is 7·9 sq. in., and the load is 7·9 × 97 = 766 lbs., hence the load per foot is $\frac{766}{5 \cdot 5} = 139 \cdot 2$ lbs. per foot.

It is usual also to include a quantity representing the free bending-moment in each bay—*i.e.*, $\frac{wl^2}{8} = \frac{139 \cdot 2 \times 5 \cdot 5^2}{8} =$ 527 lbs. ft., due to air forces which are assumed to be uniform in any one bay.

OVERHANG BENDING-MOMENT

Dealing with the overhang portion at the outer end of the spar, the bending-moment is obtained by a double integration of the load curve (i.e., $\iint w\,dx = M$).

This is much more easily done by integrating graphically in two stages i.e., from the load intensity curve a diagram of shear is evolved, and integrating the shear diagram in turn a diagram of bending-moments is produced. Any suitable scales may be chosen for drawing these diagrams, but the scales should be made clear. The notation to be adopted for bending-moments is that hogging bending-moments which occur in normal flight due to upward air load, are positive; sagging bending-moments being negative.

BENDING-MOMENT SCALE

In the case of the front spar 1 sq. in. of load intensity diagram = 1 in. on shear diagram (ordinate) and 1 sq. in. of shear diagram = 1 in. on bending-moment diagram (ordinate); hence the scales are—

$$\begin{aligned}
\text{1 sq. in. load diagram} &= \text{linear scale} \times \text{load scale.} \\
&= 2{\cdot}3\text{ ft.} \times 42{\cdot}1. \\
&= \text{1 in. ordinate on shear diagram.} \\
\text{1 sq. in. shear diagram} &= \text{linear scale} \times \text{shear scale.} \\
&= 2{\cdot}3\text{ ft.} \times (2{\cdot}3 \times 42{\cdot}1) \\
&= 42{\cdot}1 \times 2{\cdot}3^2 = 222{\cdot}5\text{ lbs. ft.} \\
&= \text{1 in. ordinate on bending-moment diagram.}
\end{aligned}$$

Thus, the maximum value of the overhang bending-moment on the top front spar is found to be 2·4 × 222·5 = 534 lbs. feet, as will be observed from Fig. 5, and the sign of this B.M. is negative.

The upper portion of this load diagram on Fig. 5 has been worked out in a similar manner and represents the case for the top rear spar with C.P. at ·3 of the chord; this, however, is not the loading which decides the size of the rear spar, but the reason why the rear spar has been worked out for this case is that it will be useful later in estimating the drag forces which depend upon the lift loads.

It will be observed that although the intensity of load may vary slightly across any one bay, yet it is assumed to be uniform in that bay, a glance at the diagram will show that this assumption is warranted.

The positions of the interplane struts are marked as 1, 2, 3, corresponding to the outer, inner, and body struts, and these are assumed to be collinear points along a continuous spar, being points where the spar is assumed supported.

FREE REACTIONS

The free reactions at the points of support can be found direct from the load diagram, but these reactions will be modified because of the fixing moments (*i.e.*, bending-moments at the supports); they will, however, be required, and are found herewith. The free reaction at the outer strut (point 1) is that due to the load on the overhang added to half the load on the spar in the bay between points 1 and 2, viz.—

$$347 + \frac{766}{2} = 730 \text{ lbs.} \quad (\textit{See } \text{Fig. 5.})$$

Similarly, the free reaction at the inner strut (point 2) is due to half the load in each of the bays adjacent to it; viz.—

$$\frac{766}{2} + \frac{630}{2} = 698 \text{ lbs., and so on for point 3.}$$

LOAD DIAGRAM FOR BOTTOM FRONT SPAR

Before going further, it is necessary to draw the load diagram for the bottom front spar. It will be noticed that in the bottom wing there is no loaded centre portion, as was the case on the top wing; furthermore, it is hinged to the fuselage at the two points of attachment. The diagram for the bottom wing is set out in Fig. 6 in exactly the same way as for the top front spar, with somewhat similar divisions for the bays between the points of support. From previous work the mean net load per square foot on the bottom wing is 5·43 lbs. per square foot. Hence the load on half the bottom wing at five times loading is = factor × net load per square foot × area of = one plane

$$= 5 \times 5{\cdot}43 \times \frac{160}{2} = 2{,}172 \text{ lbs.}$$

The total area of this load curve = 26·81 sq. ins., therefore 1 sq. in. represents $\frac{2172}{26{\cdot}81} = 81$ lbs., and the area of the lower portion of the load curve representing the load on bottom front spar = 18·94 sq. ins., hence the load on this spar is

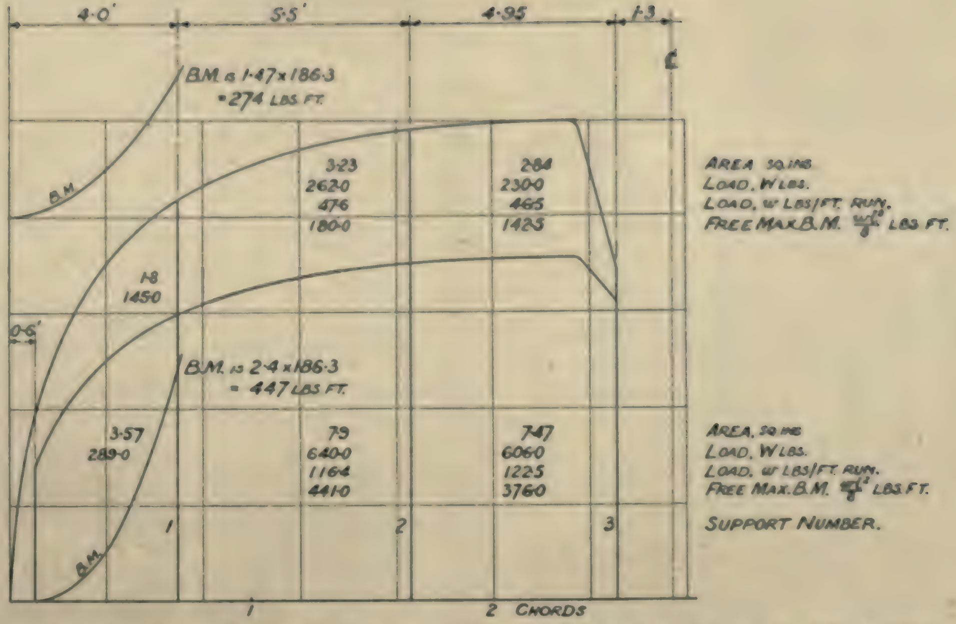

FIG. 6.—LOAD CURVE FOR BOTTOM FRONT SPAR, C.P. AT ·3 OF THE CHORD
5 TIMES LOAD

$18{\cdot}94 \times 81 = 1535$ lbs. The mean height of spar load diagram $= \dfrac{18{\cdot}94}{6{\cdot}02} = 3{\cdot}15$ ins., and the mean load per foot on spar =

$$\frac{\text{Load on spar}}{\text{Length of spar}} = \frac{1535}{13{\cdot}85} = 110{\cdot}8 \text{ lbs. per ft. run ;}$$

hence the load scale is—

$$1 \text{ in.} = \frac{\text{load per foot run}}{\text{mean height of diagram}}$$

$$\therefore 1 \text{ in. represents } \frac{110{\cdot}8}{3{\cdot}15} = 35{\cdot}2 \text{ lbs. per ft. run ;}$$

whilst the linear scale is—

$$1 \text{ in.} = 2{\cdot}3 \text{ ft. ;}$$

consequently the bending-moment scale for the overhang

$$1 \text{ in.} = 2{\cdot}3 \times 2{\cdot}3 \times 35{\cdot}2 = 186{\cdot}3 \text{ lbs. ft.}$$

The bending-moment (maximum) at the outer strut, due to load on overhang

$$= 186{\cdot}3 \times 2{\cdot}4 = 447 \text{ lbs. ft.,}$$

the sign being negative as for the top front spar.

OFFSET BENDING-MOMENTS

It is one of the ideals of every designer to design spars without offsets at the supports, for these, when present, introduce bending-moments on the spars of magnitudes too large to be neglected. In practice, however, it is often difficult to avoid an offset wire or strut, and so the spar must be made strong enough to withstand the extra bending-moment due to the offset.

The offset at each outer top joint wire attachment both front and rear is, in these calculations, ·25 in., and the wire offset at each inner top joint is ·1 in. (*see* Fig. 7) ; consequently, whatever load is taken by a lift wire it will produce a bending-moment on the top spars in the bay affected. It is now necessary to estimate these bending-moments in order to solve the three moment equations, and for this purpose a diagram (Fig. 8) is drawn, not necessarily to scale, showing the front truss of the wings and containing a few principal dimensions, stagger and dihedral of the wings being neglected for the present.

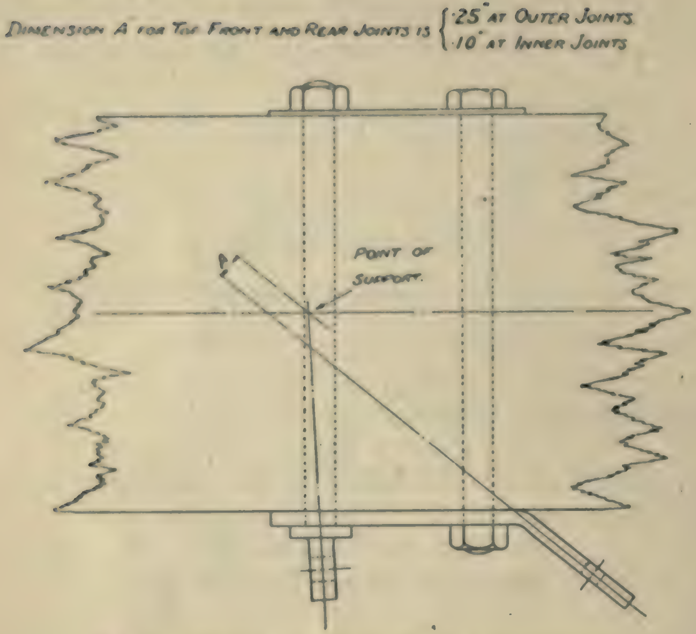
Dimension A for Top Front and Rear Joints is { ·25" at Outer Joints
·10" at Inner Joints
Point of Support.
A

Fig. 7.—Strut and Lift Wire Attachments

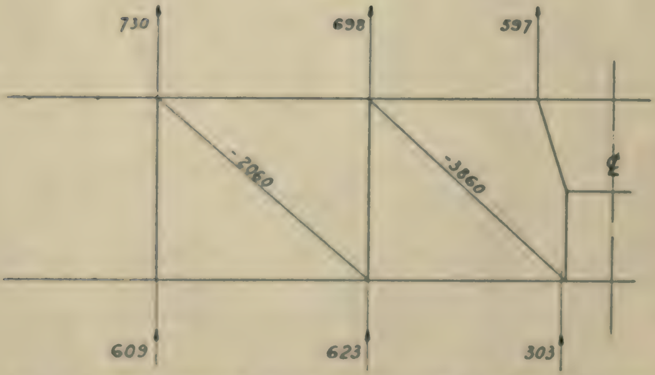

Fig. 8.—Preliminary Free Reactions for Front Truss
C.P. at ·3 of the Chord, 5 Times Load

The reactions shown in the diagram (Fig. 8) are free reactions obtained direct from the load curves for top and bottom wings, front spars, and the immediate object is to estimate the loads in the lift wires, and so approximate to the offset bending-moment produced. The method of obtaining loads in spars and bracing is explained on p. 37. The offset bending-moment at the top outer joint

$$= 2060 \times \frac{\cdot 25}{12} = 43 \text{ lbs. ft., say, } 40 \text{ lbs. ft.}$$

and the offset bending-moment at the top inner joint

$$= 3860 \times \frac{\cdot 1}{12} = 32 \text{ lbs. ft., say, } 30 \text{ lbs. ft.}$$

The offsets are such that the offset bending-moment is positive in the outer and inner bays respectively.

PRELIMINARY REACTIONS, Top Front Spar

It is now possible to determine the modified reactions at the supports by solving the three moment equations, but these will be approximate, inasmuch as end load in the spar is neglected at present, but will be taken into account later.

From the load diagram top front spar the overhang bending-moment is seen to be -534 lbs. ft., consequently, at the inner side of the outer support the bending-moment is

$$-534 + 40 = -494 \text{ lbs. ft.,}$$

taking account of the offset bending-moment there.

The theorem of three moments for the spar is—

$$M_1 l_1 + 2(l_1 + l_2)M_2 + M_3 l_2 + \frac{6A_1\bar{x}_1}{l_1} + \frac{6A_2\bar{x}_2}{l_2} = 0$$

$$\text{or } M_1 l_1 + 2(l_1 + l_2)M_2 + M_3 l_2 = -\frac{6A_1\bar{x}_1}{l_1} - \frac{6A_2\bar{x}_2}{l_2}$$

$$= -2\,m_1 l_1 - 2\,m_2 l_2$$

Where A = Area of applied B.M. diagram.

$\bar{x}$ = distance of C.G. of B.M. diagram from point of support.

m = free maximum B.M. due to wind load in bay concerned.

l = length of bay.

It should be noted that $\frac{6A\bar{x}}{l}$ becomes $2\,ml$ in the case of a parabolic bending-moment due to uniform air load.

and also that for a triangular offset bending-moment it becomes 2 *nl* or *nl* as the case may be, where n = maximum bending-moment due to offset load.

Let the fixing moments on the outer side (*i.e.*, the left-hand side in the diagram) of the points of support 1, 2, and 3 to be M_1, M_2, and M_3 respectively.

Hence, setting down the equations for the spar, including offset bending-moments—

$$-494 \times 5{\cdot}5 + 2(5{\cdot}5 + 4{\cdot}25)M_2 + 4{\cdot}25\,M_3 = -2 \times 5{\cdot}5 \times 527 - 2 \times 4{\cdot}25 \times 335 - 2 \times 30 \times 4{\cdot}25$$

from which

$$19{\cdot}5\,M_2 + 4{\cdot}25\,M_3 = -6185 \quad . \quad . \quad . \quad (1)$$

$$4{\cdot}25\,M_2 + 2(4{\cdot}25 + 4)M_3 + 4M_3 = -2 \times 4{\cdot}25 \times 335 - 2 \times 4 \times 282 - 30 \times 4{\cdot}25$$

from which

$$4{\cdot}25\,M_2 + 20{\cdot}5\,M_3 = -5233 \quad . \quad . \quad . \quad (2)$$

and, solving (1) and (2)—

$$M_2 = -274 \text{ lbs. ft. and } M_3 = -198 \text{ lbs. ft.}$$

The modified reactions will approximate to the final reactions and are—

$$\begin{aligned} R_1 &= 347 + 383 + \frac{494 - 274}{5{\cdot}5} &&= 772 \text{ lbs.} \\ R_2 &= 383 + 315 - 42 + \frac{274 - 198}{4{\cdot}25} &&= 660 \text{ lbs.} \\ R_3 &= 315 + 282 - 4 &&= 593 \text{ lbs.} \\ & &&\quad \overline{2025 \text{ lbs.}} \end{aligned}$$

We must now carefully estimate the bending-moments at the various points of support, and thence the corresponding modified reactions for the bottom front spar exactly as was done for the front top spar.

Bottom Front Spar

It is assumed that there are no offset wire attachments on the bottom spars, and that all the wires pass through the points of support, consequently there are no offset bending-moments at those points. It will be noticed that the bottom wing is hinged at its points of attachment to the fuselage, hence M_3 on the bottom spars = 0, and it remains to find M_2 by solving the equation.

The bending-moment at the overhang on the bottom front spar $= -447$ lbs. ft.

$$\therefore\ -447 \times 5{\cdot}5 + 2(5{\cdot}5 + 4{\cdot}95)M_2 + 0 = -2 \times 441 \times 5.5 - 2 \times 376 \times 4{\cdot}95$$

from which

$$20{\cdot}9\ M_2 = -4850 - 3720 + 2460 = -6110$$

hence $M_2 = -292$ lbs. ft.

The modified reactions are, therefore—

$$\begin{aligned} R_1 &= 289 + 320 + \frac{447 - 292}{5{\cdot}5} &&= 637 \text{ lbs.} \\ R_2 &= 320 + 303 - 28 + \frac{292}{4{\cdot}95} &&= 654 \text{ lbs.} \\ R_3 &= 303 - 59 &&= \underline{244 \text{ lbs.}} \\ & && \quad \underline{\underline{1535 \text{ lbs.}}} \end{aligned}$$

These modified reactions will be found to differ somewhat from the former free reactions, due to the bending-moments at the supports, although the total remains the same, of course.

Since the loads in the drag bracing impose end loads in the spars which add to the end loads due to lift, and since the drag is some proportion of the lift reaction at each joint, then it is essential that the reactions on the spars, both front and rear, must be obtained. This means that the modified reactions on the rear spars, top and bottom, must be calculated in exactly the same way as has been done for the top and bottom front spars.

Top Rear Spar : Centre of Pressure Forward

From the upper portion of the load curve in Fig. 5 the necessary particulars relating to the rear top spar are readily obtainable. The offsets of the lift wires are ·25 in. and ·1 in. for the outer and inner lift wires respectively, and the load in rear outer lift wire for this case = 932 lbs. ; therefore, offset

B.M. at point 1 $= 932 \times \frac{\cdot 25}{12} = 20$ lbs. ft.

The load in rear inner lift wire = 1650 lbs., therefore, offset B.M. at point 2 $= 1650 \times \frac{\cdot 1}{12} = 14$, say, 15 lbs. ft.

The overhang bending-moment from Fig. 5 = − 327 lbs. ft. and the equations are (including offset B.M.'s)—

$$-307 \times 5{\cdot}5 + 2(5{\cdot}5 + 4{\cdot}25)M_2 + 4{\cdot}25\,M_3 = -2 \times 5{\cdot}5 \times 214 - 2 \times 4{\cdot}25 \times 136 - 2 \times 15 \times 4{\cdot}25$$

for outer and inner bays, from which

$$19{\cdot}5\,M_2 + 4{\cdot}25\,M_3 = -1952 \quad . \quad . \quad . \qquad (1)$$

and

$$4{\cdot}25\,M_2 + 2(4{\cdot}25 + 4)M_3 + 4M_3 = -1155 - 2 \times 4 \times 63 - 15 \times 4{\cdot}25$$

from which

$$4{\cdot}25\,M_2 + 20{\cdot}5\,M_3 = -1723$$

hence $M_2 = -86$ lbs. ft. and $M_3 = -66$ lbs. ft.

The modified reactions, which will approximate to the final reactions are—

$$\begin{array}{lll} R_1 = 174 + 156 + \dfrac{307 - 70}{5{\cdot}5} & = & 373 \text{ lbs.} \\ R_2 = 156 + 128 - 43 + \dfrac{4}{4{\cdot}25} & = & 242 \text{ lbs.} \\ R_3 = 128 + 63 - 1 & = & 190 \text{ lbs.} \\ & & \overline{805 \text{ lbs.}} \end{array}$$

Bottom Rear Spar : Centre of Pressure Forward

Similarly, from Fig. 6 particulars are obtainable from which the equations can be framed to find M_2, M_3 being zero on account of the hinge at 3.

The maximum value of the overhang bending-moment is − 274 lbs. ft., and offsets are zero ; therefore,

$$-274 \times 5{\cdot}5 + 2(5{\cdot}5 + 4{\cdot}95)M_2 + O = -2 \times 5{\cdot}5 \times 180 - 2 \times 4{\cdot}95 \times 142{\cdot}5$$

$$20{\cdot}9\,M_2 = -1980 - 1412 + 1505 = -1887\,;$$

hence, $M_2 = -90$ lbs. ft., and the modified reactions, which will approximate to the final reactions, are—

$$\begin{array}{lll} R_1 = 145 + 131 + \dfrac{274 - 90}{5.5} & = & 309 \text{ lbs.} \\ R_2 = 131 + 115 - 33 + \dfrac{90}{4{\cdot}95} & = & 231 \text{ lbs.} \\ R_3 = 115 - 18 & = & 97 \text{ lbs.} \\ & & \overline{637 \text{ lbs.}} \end{array}$$

PRELIMINARY BENDING-MOMENT DIAGRAM

Having now determined the bending-moments at the several points of support along the spars, and already knowing the offset bending-moments at these points, and the bending-moments in the bays due to uniform air forces, it is quite possible to draw a preliminary bending-moment diagram for the complete top spar, since the overhang bending-moment curve may be copied directly from the load diagram, Fig. 5.

This diagram, when drawn, will enable us to make a very good estimate of the size of spar section likely to be suitable, because it enables us to see at a glance what is the maximum bending-moment along the spar.

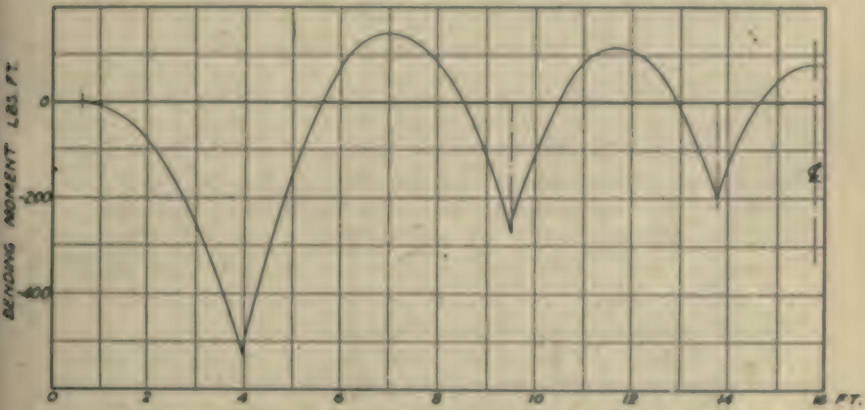

Fig. 9.—Preliminary B.M. Diagram. Top Front Spar

For this case of centre of pressure forward, only two preliminary bending-moment diagrams need be drawn (*i.e.*, one for the top front spar and the second for the bottom front spar) because this is not the case which determines the size of the rear spars. The case in which the maximum load is taken by the top and bottom rear spars is that when the centre of pressure is at ·5 of the chord, and that is the case which determines the sizes of those spars.

Dealing first with the top front spar, the points of support are set out to scale, and the bending-moments at those points marked off to some scale.

In the example, the maximum overhang bending-moment from Fig. 5 is : – 534 lbs. ft. ; this is marked off at point 1 on Fig. 9. The effect of the offset bending-moment is such as to help the hogging bending-moment within the bay 1 to 2, and therefore, according to our convention of signs, will have the effect of decreasing the bending-moment immediately on the inside of the outer support. The bending-moment just inside the outer support is, then—

$$-534 + 40 = -494 \text{ lbs. ft.}$$

At the inner support point 2, the bending-moment has been found to be – 274 lbs. ft., and since the offset bending-moment is again helping the hogging effect of the air lift forces, the bending-moment just to the inside of this support will be reduced by 30 lbs. ft. ; *i.e.*—

$$-274 + 30 = -244 \text{ lbs. ft.}$$

Regarding the support point 3, there is no offset bending-moment, and the fixing moment there has already been found to be – 198 lbs. ft. These B.M.'s must now be set out in Fig. 9 and the curves inserted. In the case of the overhang, this curve may be assumed parabolic and need not be copied directly from the load curve Fig. 5 ; also regarding the B.M. curves in the other bays, these may be plotted as parabolic in this preliminary diagram, which is only constructed in order to make a reasonable estimate as to the spar section.

In each bay the parabola must be plotted on the line as base which joins the two points representing the magnitudes of bending-moments at the supports of that bay. (*See* Fig. 9.) Of the several methods of drawing a small portion of a parabola quickly and accurately, the following is probably the most convenient.

Divide up the vertical depth into four equal parts, and draw lines through these divisions to cut the side boundaries each into four points, now divide the base line into eight equal parts and drop a perpendicular through each point. Join the apex to the several divisions on the side boundaries to cut the verticals in various points. It will be quickly observed from Fig. 9 how this has been done, and which are the points forming the curve when connected by an even line.

The preliminary B.M. diagram is now completed, and it is an easy matter to see exactly which point gives maximum bending-moment.

Actually, the spar will be solid at the points of support, but we may neglect that at present and will take account of it later.

The point we have found giving maximum bending moment may not be the point where maximum stress is induced in the spar, because very often direct end load compressive stress $\left(\frac{\text{End load}}{\text{area}}\right)$ plays a very important part in this respect.

It is not intended to attempt to obtain any idea of the end load bending-moments at present, for these will be automatically included in the more accurate method to be applied later.

It would be a very tedious business to do so, for it would mean two graphical integrations of the B.M. diagram in order to find a deflection diagram, and there is all the difficulty of making the deflection curve pass through the several points of support, which points must be in alignment. Having done this, the end load B.M. would be found by taking the product of the end load and the deflection in each bay.

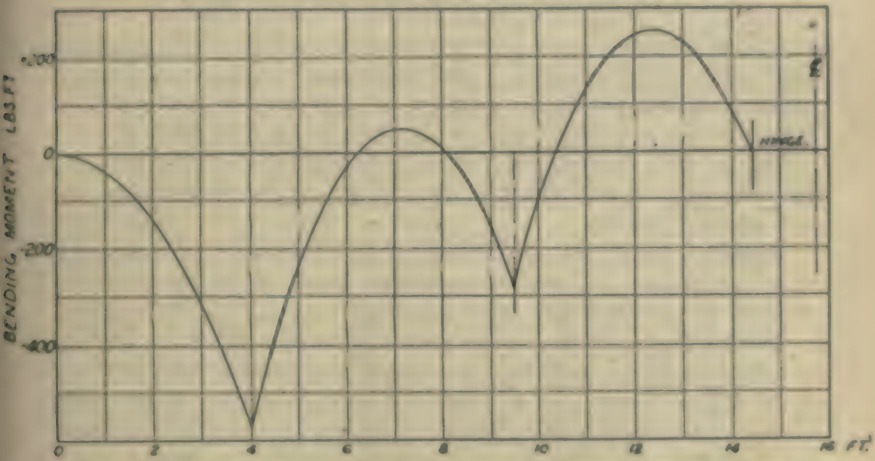

FIG. 10.—PRELIMINARY B.M. DIAGRAM. BOTTOM FRONT SPAR

We shall need to obtain the end loads in the spars in any case, and to do this a diagram of struts and wires must be made out.

Following similar lines for the bottom front spar as for the top front spar, a preliminary bending-moment diagram may be constructed by obtaining the required particulars from the load diagram (Fig. 6), the bending-moments at the supports having been found previously from the three moment equations which were solved for the bottom front spar.

It is noteworthy that this is an example of a spar in which no offset bending-moments occur, and it should be considered as exceptional in that respect, as in actual practice it is difficult to avoid them.

Fig. 10 shows the diagram completed, and it should be noted that no centre portion exists on the bottom plane in this example since the spars merely attach to the fuselage by means of hinges, hence the bending-moment at the point of attachment to the fuselage is zero and the diagram is more simple in that respect.

DIAGRAM OF STRUTS AND WIRES

These modified reactions may now be set out on a diagram (not necessarily to scale), of the front and rear trusses, the rear truss being shown at the bottom of the diagram, the plans of top and bottom drag bracing also being shown. (Fig. 11.)

Before this diagram can be completed the drag forces must be determined, and drag is usually taken as one-seventh the lift at any point.

Account must also be taken of the direction of lift wires, and the stagger of the wing struts, both of which generally modify the drag forces on the wings.

The attitude of the aeroplane in this case of C.P. forward is assumed to be such that the angle of incidence of the wings is 16 degrees, and it is advisable that a side view of the wing chord, struts, and wires should be drawn to suit this condition, as is shown in Fig. 12, the direction of lift being taken perpendicular to the line of flight.

DRAG FORCES

Commencing with the bottom end of the front outer strut, the reaction is 637 lbs., and so, in order to estimate the drag there, a vertical line is drawn to scale to represent this reaction, and a line perpendicular to it of one-seventh the length, to represent the drag. (*See* Fig. 13.) The resultant lift is then the sum of these two component forces, and the resultant is taken partly up the strut and partly along the wing from front to rear. The lift reaction at top of front outer strut = 772 lbs., and the resultant lift is found in exactly the same way as before. The forces at this point are resultant lift, thrust up

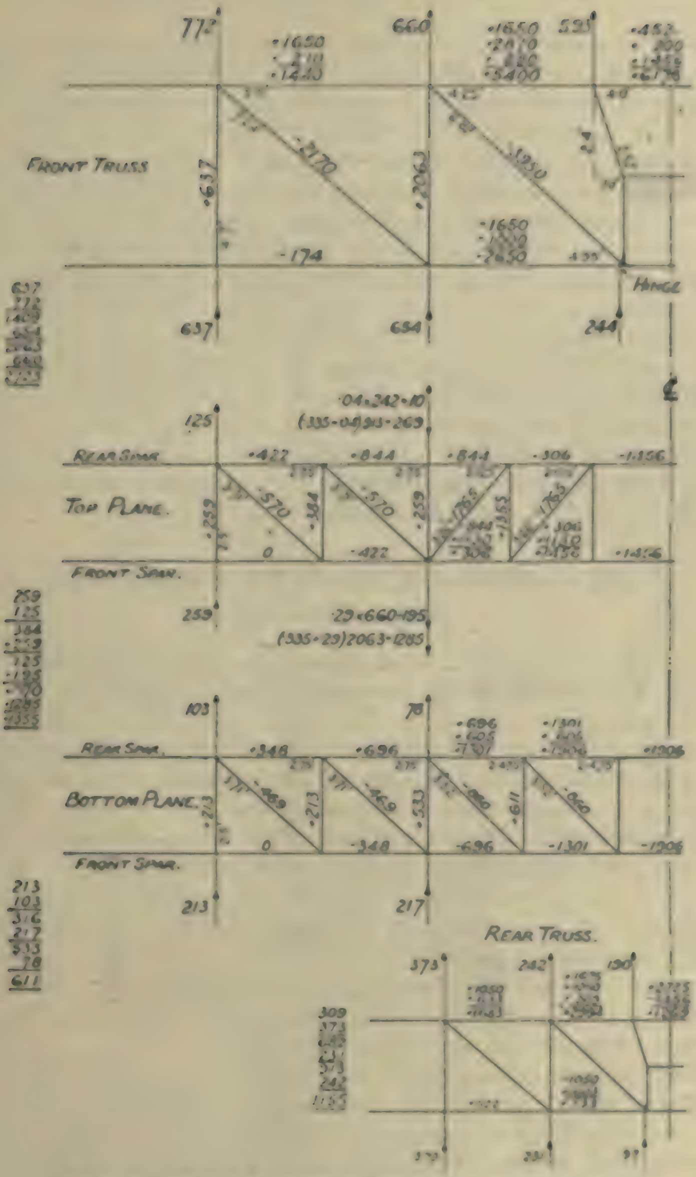

FIG. 11.—FORCE DIAGRAM FOR FRONT TRUSS. C.P. AT ·3 OF THE CHORD. 5 TIMES LOAD

the outer strut due to lift from bottom wing, drag force along wing from front to rear, and tension in outer lift wire. These forces are represented in the vector diagram shown (Fig. 13), and the drag force from front to rear obtained.

A similar procedure at the front bottom inner joint enables the drag force to be determined there. A little care must be exercised at the top inner strut joint, where the main lift wire is not in the same plane as the strut but is forward of it, and since the load in this wire is very large, it has the effect of producing a forward drag force there. The forces at the joint are: resultant lift, thrust up the strut, load in main lift wire, and drag force. The thrust up the strut is obtained from the vector force diagram of the lower inner joint. It is not immediately necessary to estimate the drag forces at the body struts, since these do not affect the bracing in the wings. The drag forces for the rear truss C.P. forward must, however, be calculated in order to complete Fig. 11.

ALTERNATIVE METHOD OF ESTIMATING DRAG FORCES

An alternative method of finding drags, and one which is more convenient than the one already described, is the following—

Only one diagram need be drawn, for it will be seen that this is sufficient for all the joints. In setting out the diagram, draw a horizontal line representing the line of flight, and a line perpendicular to it representing the lift 7 units long, drag being 1 unit long horizontally, as shown in Fig. 14, which enables the direction of resultant lift to be drawn on the diagram. From a point on the horizontal flight path, set out the directions of front and rear, outer and main lift wires, struts, and incidence wires, and through the same point draw the chord of the bottom wing, making an angle of 16 degrees (angle of incidence) above the horizontal.

Let the resultant lift OA be marked unity, and draw the chord of the top wing at the same angle of incidence, which will cut the lines drawn parallel to lift wires, struts, etc., in various proportions, shown as OB, OC, OD, OE. The drag forces on the wings may now be calculated by consideration of the separate joints, commencing with the bottom front outer joints where the lift is 637 lbs. As previously stated, the

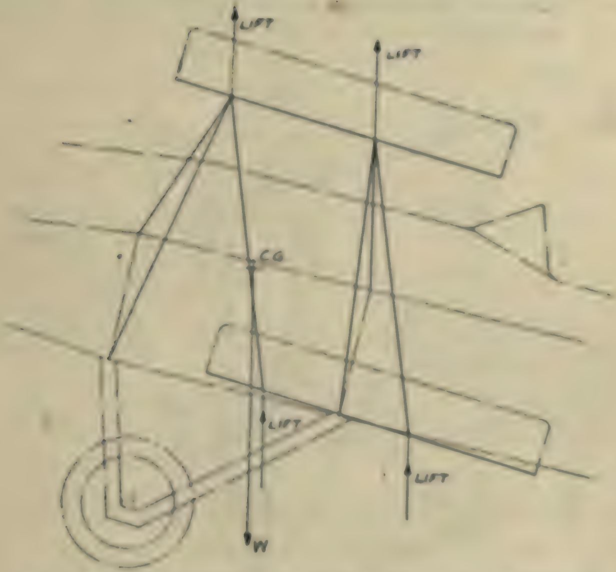

FIG. 12.—ATTITUDE OF AEROPLANE WHEN CLIMBING AT AN ANGLE OF INCIDENCE OF 16 DEGREES

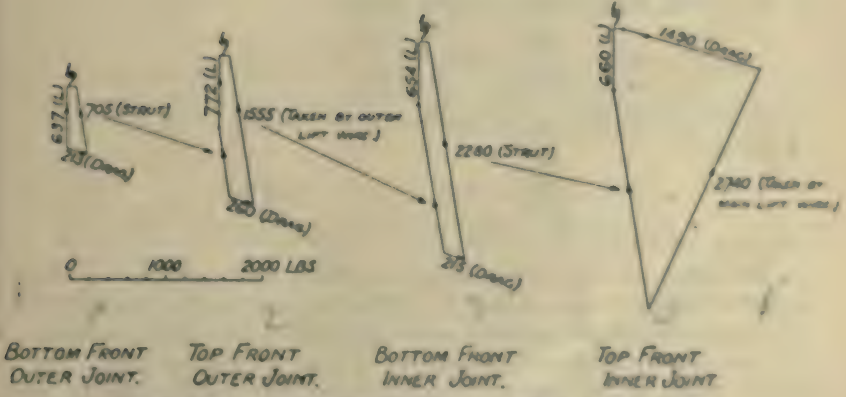

FIG. 13.—DRAG COEFFICIENT DIAGRAM, C.P. AT ·3 OF THE CHORD

forces at that joint are: resultant lift, thrust up the strut, and drag force on the wings, so that OAB is the triangle of forces for the joint.

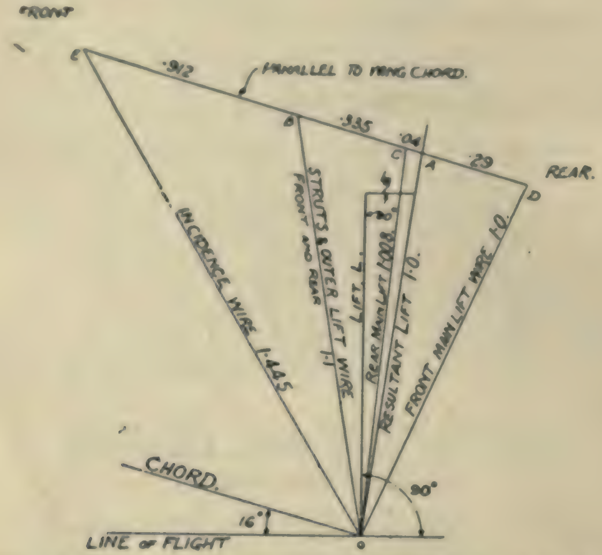

FIG. 14.—DRAG COEFFICIENTS DIAGRAM C.P. AT ·3 OF THE CHORD. ANGLE OF INCIDENCE 16 DEGREES

The load taken by the strut is thus—

$$= 637 \times \frac{OB}{OA} = 637 \times \frac{1 \cdot 1}{1 \cdot 0} = 702 \text{ lbs.}$$

and the drag force on wings, which is taken by the drag bracing, is

$$= 637 \times \frac{AB}{OA} = 637 \times \cdot 335 = 213 \text{ lbs.}$$

from front to rear.

The next joint in order is the top front outer joint (lift 772), where the forces are: load up strut, resultant lift, drag force on wing, and the tension in outer lift wire. The force diagram is, therefore, OAB as before, since the outer lift wire is in the same plane as the strut, which means that no drag force is imposed on the wing by the transference of load from strut to wire.

Hence, the drag load on wing

$$= 772 \times \frac{AB}{OA} = 772 \times {\cdot}335 = 259 \text{ lbs.}$$

in a direction from front to rear.

Bottom Front Inner Strut Joint

Here again the triangle of forces is the same, and drag force on the wing

$$= {\cdot}335 \times \text{lift} = {\cdot}335 \times 654 = 217 \text{ lbs.}$$

from front to rear.

Considering the top front inner joint, where the lift = 660 lbs., the forces are: resultant lift, load up the strut, drag force on the wing, and tension in lift wire, which is not in the same plane as the struts; hence the force diagram at the joint is no longer a triangle.

It is convenient, however, to consider the local lift at the joint separately from the lift transferred from the other points already considered. This latter is taken up the strut, in which case there are two triangles of forces; namely, *OAD* and *OBD* respectively.

The total drag force on the wing will therefore be negative and

$$= \frac{AD}{OA} \times 660 + \frac{BD}{OA} \times (637 + 772 + 654)$$

$= 660 \times {\cdot}29 + 2063 \times ({\cdot}335 + {\cdot}29) = 1285$ lbs. from rear to front, as shown in Fig. 11.

In a similar manner, the drag forces for the rear truss C.P. forward are determined by using the reactions found for the rear joints, shown in Fig. 11, the directions being seen by inspection.

LOADS IN SPARS AND BRACING

All the drag forces thus calculated are now inserted at their respective joints in Fig. 11, and the loads in the spars and bracing obtained either graphically or by calculation, but it is generally done more quickly and probably more accurately by calculation. For this purpose all the lengths of the wires and the distances between points of support along the spars are required, and the loads are then obtained by the proportion of the lengths of the triangles formed; for example, in the front truss the load in the outer lift wire (neglecting stagger) is

$$(637 + 772)\frac{7{\cdot}24}{4{\cdot}7} = 2170 \text{ lbs.},$$

and the compressive load in the top spar outer bay due to lift load only, is the horizontal component of this load; *i.e.*,

$$2170 \times \frac{5{\cdot}5}{7{\cdot}24} = 1650 \text{ lbs.}$$

this latter can also be obtained direct; *i.e.*—

$$(637 + 772)\frac{5{\cdot}5}{4{\cdot}7} = 1650 \text{ lbs.}$$

In a similar manner the loads in the remainder of the truss, due to the lift loads only, can readily be calculated, and these also are marked on the diagram. (Fig. 11.) End loads in the spars are also caused by the drag forces. These end loads due to drag are easily calculated for the bottom wing, where all the drag forces are backward, but in the top wing there is a reversal of direction at the inner strut. A little care must be exercised there, but in any case, however, the work can be checked graphically. The end loads due to drag in the front spars must now be transferred to the front truss diagram, and added to the end loads due to lift. It will be noticed, however, that in some bays the end load due to drag differs in one portion of the bay from another, but the mean of the two loads is considered to be quite reliable.

Thus in the top front bay the end load due to drag is O and − 422 lbs. in the two halves of the bay, having a mean of − 210 lbs., which is to be added algebraically to the 1650 end load due to lift. In the inner bay the net end load is that due to the main lift wire component (2870 lbs.), plus the already existing end load (1650 lbs.) due to the outer lift wire, and to these is still to be added the mean end load of 880 lbs. due to drag forces on the top wing, making a total compressive end load of 5400 lbs.

In the centre portion of the top spar the end load due to the body strut must also be taken into account.

PRELIMINARY SECTION OF FRONT SPARS

Although the exact section at every point of the spar cannot yet be decided upon, since we do not know what will be the final bending-moments and end loads everywhere along the spar, we can, by taking account of the work already done, get some idea of the probable size of spar suitable to the final figures.

The depth, of course, is fixed by the wing section,* but the width, and the portions to be spindled, are at the disposal of the designer.

A rough trial section for the top front spar is now to be assumed having rectangular flanges 1·5 in. wide × ·35 in. thick, rectangular web ·35 in. thick, the total depth of section about 4 in.

An approximate idea of the section modulus of this section is given by : $\frac{BD^3-bd^3}{6D}$ where BD, bd, are maximum and minimum breadths and depths respectively ; thus—

$$Z = \frac{1{\cdot}5 \times 4^3 - 1{\cdot}15 \times 3{\cdot}3^3}{6D} = 2{\cdot}3 \text{ nearly.}$$

The area of the section = 2·2 sq. in.

Approximate stress in the spar at support 3 just inside the centre bay (see Fig. 9) is—

B.M. stress $= \frac{198 \times 12}{2{\cdot}3} = 1035$ lbs. per sq. in.

Direct end load stress $= \frac{6176}{2{\cdot}2} = 2810$

Total stress = 3845 lbs. per sq. in.

Allowable stress = 5500 lbs. per sq. in.

Thus it would appear that the spar section is certainly on the strong side.

It ought still to be remembered, however, that we have neglected stress due to end load bending-moment, and in any case the spar section is such that it would not be wise from practical considerations and a manufacturing point of view to make any further reduction in the width or thickness.

The spar section for bottom front spar is assumed to be the same as for the top front spar (*i.e.*, $Z = 2{\cdot}3$), and the area of the section $A = 2{\cdot}2$ sq. ins.

If the stress due to bending-moment and due to direct end load be found at one or two points, it will be seen that the spar is up to strength.

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* The aerofoil section known as R.A.F. 15 is here assumed in the example.

CHAPTER III

CENTRE OF PRESSURE AT ·3 OF CHORD

MOMENT OF INERTIA AND SECTION MODULUS

HAVING now concluded that the trial spar section is of about the right order as regards overall dimensions, it now remains to draw out what is anticipated will be the final section, and find as accurately as possible its moment of inertia, area, and the section modulus. This anticipated section is shown in Fig. 15. It is assumed that no spindling of the spar commences within 3 in. from the points of support, and furthermore, that the spar is drilled vertically at the points of support to accept a 5/16 in. diameter B.S.F., vertical bolt.

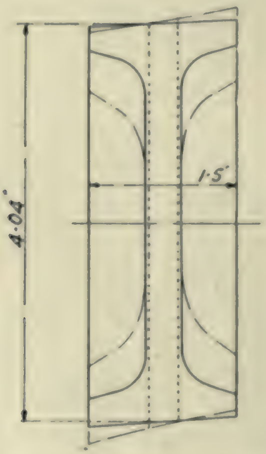


FIG. 15.—FRONT SPARS; MOMENT OF INERTIA DIAGRAM

Thus, there is a solid portion extending for 3 ins. on either side of each support, which is essential for receiving fittings to which are attached struts and lift wires.

There are several methods of finding the moment of inertia of a section such as is shown, but probably the most convenient method is a graphical one, and of all the graphical ones, probably the best of them is the one about to be described.

Plate 2

Austin "Greyhound" with 320 H.P. A.B.C. "Dragonfly" Radial Engine

Bristol Fighter F.2.B. Aeroplane with Rolls-Royce Engine

F.E.4 Aeroplane with 2-150 H.P. R.A.F. Engines

Most of the others are difficult to use, and liable to inaccuracy, whereas this one can be recommended for its simplicity and general reliability. It is an old method and it is difficult to say who originated it ; in any case it is not so generally known as it ought to be, considering its usefulness. The method is as follows—

Find the neutral axis of the section, then cube the length of the ordinates (it will be found an advantage to use a table of cubes for this purpose) with the neutral axis as base ; set out this cubed length along its own ordinate, and draw the outline of the figure thus formed.

The moment of inertia of the whole section is then equal to the area of this cube figure, on either side of the neutral axis divided by 3.

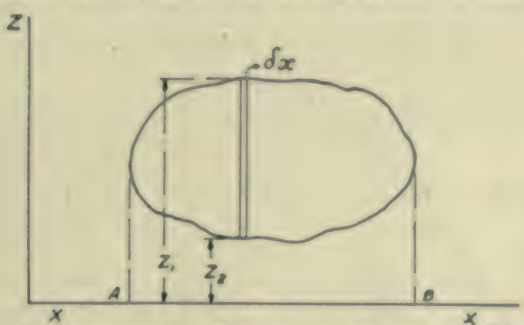


FIG. 16.—DIAGRAM TO ILLUSTRATE PROOF OF METHOD TO FIND MOMENT OF INERTIA

The modulus of the section is then easily found by dividing by half the depth of the section. If it is necessary to draw the cube figure to scale, then this must be accounted for in the calculation of I.

Moment of Inertia (Proof of Rule)

Proof of method described to find I of any figure.

The moment of inertia of a figure about any axis is

$$I_{xx} = I_o + A \times d^2$$

where

I_o = moment of inertia about an axis through centre of gravity.
I_{xx} = moment of inertia about new axis.
A = area of section considered.
d = perpendicular distance between the two axes.

Referring to Fig. 16—

Consider any strip dx in width, then

$$I_o = \frac{\text{breadth} \times (\text{depth})^3}{12}$$

$$= \frac{dx\,(z_1 - z_2)^3}{12}$$

Then I_{xx} may be written as

$$= \frac{(z_1 - z_2)^3\,dx}{12} + (z_1 - z_2)\,dx\left(\frac{z_1 + z_2}{2}\right)^2$$

$$= \frac{1}{12}\Big[(z_1 - z_2)^3\,dx + 3\,(z_1^2 - z_2^2)\,(z_1 + z_2)\,dx\Big]$$

Then moment of inertia of whole area is

$$I_{xx} = \frac{1}{12}\int_A^B \Big[(z_1 - z_2)^3 + 3(z_1^2 - z_2^2)\,(z_1 + z_2\Big]dx$$

which reduces to

$$I_{xx} = \tfrac{1}{3}\int_A^B (z_1^3 - z_2^3)\,dx$$

Fig. 15 shows the hollow spar section, also the solid section with $\frac{5}{16}$ in. vertical bolt, as well as the cube diagrams for both these sections, and tabulated below are given the areas of the sections themselves, the areas of the cube figures (which in this case have been drawn $\frac{1}{4}$ full size), also the values of I and Z for the sections.

FRONT SPARS

$I = \frac{A}{3} \times 4$, SINCE THE CUBE SCALE IS $\frac{1}{4}$ FULL SIZE

SECTION.	Area of Cube Fig.	I	Z	Area of Section.
	sq. in			sq. in
Solid Section . .	6·16	8·21	4·0	6·06
Solid Section ($\frac{5}{16}$ hole)	4·88	6·51	3·17	4·80
Hollow	3·58	4·77	2·32	2·3

CORRECTED REACTIONS AND B.M.'s

It is now possible to apply the more detailed and more correct method to this case of C.P. at ·3 of the chord. This method is a very close approximation to the exact mathematical solution, and is due to Messrs. Webb & Thorne, who originated it as being easily applicable to design, instead of the more cumbersome method previously worked out by Webb himself.

Webb's original method was standardized, however, for the checking of designs in the R.A.E., from 1916 onwards, and it was with the idea of making the method easily applicable to design that the latter method was evolved.

It was first published in "*Aeronautics,*" January, 1919, and a summary of the method will be found in the appendix to these pages, where the notation is made clear.

It will be found to be quite an advantage to tabulate the various quantities required in the formulae used, and so, as an example of how this ought to be done, Table I shows in tabular form the quantities necessary to the finding of the B.M.'s at the various supports, and ultimately the whole B.M. diagram for the top front spar.

FINAL REACTIONS (Top Front Spar)

The corrected bending-moments at various points in the several bays may now be calculated by considering each bay separately, and by making use of the quantities previously tabulated in Table I.

The equations for M_2 and M_3 are (from (1) in Appendix)*—

(Outer and Inter. Bays.)

$$-489 \times 1{\cdot}0166 \times 1{\cdot}14 \times 10^{-5} + 2M_{2_L} \times {\cdot}9685 \times 1{\cdot}14 \times 10^{-5} + 1054 \times {\cdot}9988 \times 1{\cdot}14 \times 10^{-5} + M_{3_L} \times 1{\cdot}037 \times {\cdot}989 \times 10^{-5} + 2(M_{2_L} + 33)\,{\cdot}9297 \times {\cdot}989 \times 10^{-5} + 670 \times {\cdot}9974 \times {\cdot}989 \times 10^{-5} = O.$$

Dividing through by $\cdot989 \times 10^{-5}$,

$$-574 + 2{\cdot}23M_{2_L} + 1{\cdot}8594M_{2_L} + 1{\cdot}037M_{3_L} = -1213 - 61{\cdot}4 - 668$$

$$\text{or } 4{\cdot}0894M_{2_L} + 1{\cdot}037M_{3_L} = -1368{\cdot}4 \quad . \quad . \quad (A)$$

(Inter. and Centre Bays.)

$$(M_{2_L} + 33) \times 1{\cdot}037 \times {\cdot}989 \times 10^{-5} + 2M_{3_L} \times {\cdot}9297 \times {\cdot}989 \times 10^{-5} + 670 \times {\cdot}9974 \times {\cdot}989 \times 10^{-5} + M_{3_L} \times 1{\cdot}0375 \times {\cdot}935 \times 10^{-5} + 2M_{3_L} \times {\cdot}9288 \times {\cdot}935 \times 10^{-5} + 564 \times {\cdot}9973 \times {\cdot}935 \times 10^{-5} = O.$$

Dividing through by $\cdot935 \times 10^{-5}$

$$1{\cdot}096M_{2_L} + 1{\cdot}965M_{3_L} + 1{\cdot}0375M_{3_L} + 1{\cdot}8576M_{3_L} = -36{\cdot}2 - 706 - 563$$

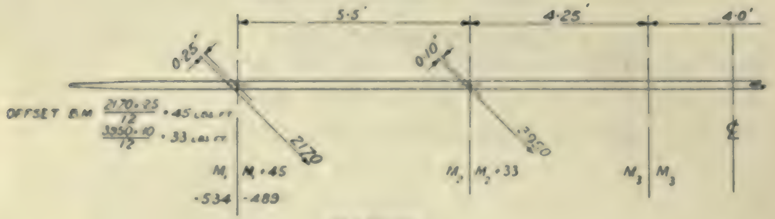
$$\text{or } 1{\cdot}096M_{2_L} + 4{\cdot}86M_{3_L} = -1305{\cdot}2 \quad . \quad . \quad . \quad (B)$$

---

* Refers to number of equation in Webb-Thorne method in Appendix

$$(A) \div 3{\cdot}73 \qquad \begin{aligned} 1{\cdot}096M_{2_L} + 0{\cdot}278M_{3_L} &= -367 \\ 1{\cdot}096M_{2_L} + 4{\cdot}860M_{3_L} &= -1305{\cdot}2 \\ \hline 4{\cdot}582M_{3_L} &= -938{\cdot}2 \\ M_{3_L} &= -204 \text{ lbs. ft.} \end{aligned}$$

$$\text{From } (B)\ M_{2_L} = \frac{-1386{\cdot}4 + 212}{4{\cdot}089} = -283 \text{ lbs. ft.}$$



5·5'
4·25'
4·0'
0·25'
0·10'
OFFSET B.M $\frac{2170 \cdot 25}{12}$ = 45 LBS FT
$\frac{3950 \cdot 10}{12}$ = 33 LBS FT
2170
3950
M_1
$M_1 + 45$
−534
−489
M_2
$M_2 + 33$
M_3
M_3
℄

TABLE 1

Span (ft.)	5·5	4·25	4·0
Span (ins.)	66	51	48
I (in.^{4})	4·77	4·77	4·77
$P_e = \frac{\pi^2 EI}{l^2}$	17400	29200	32900
End load P (comp.)	1440	5400	6176
$P_e - P$	15960	23800	26724
w (lbs. ft.)	139·2	148·3	141·0
$\frac{wl^2}{4}$ (l in ft.)	1054	670	564
$\frac{P}{P_e}$	·0828	·1850	·1875
$1 + 0{\cdot}2\frac{P}{P_e}$	1·01656	1·0370	1·0375
$1 - 0{\cdot}38\frac{P}{P_e}$	·9685	·9297	·92875
$1 - \frac{P}{70\,P_e}$	·998818	·997355	·997327
$\frac{1}{(P_e - P)\,l}$ (l in. ft.)	$1{\cdot}14 \times 10^{-5}$	$\cdot989 \times 10^{-5}$	$\cdot935 \times 10^{-5}$
M_R M_L	−489 −283	−250 −204	−204 −204
$\frac{1}{2}(M_R + M_L)$	− 386	− 227	− 204
$1 + \cdot26\frac{P}{P_e}$	1·0215	1·04815	1·0488
$1{\cdot}02\left(\frac{wl^2}{8}\right)$	538	342	287·5
$(M_L - M_R)$	+ 206	+ 46	0
$\frac{(M_L - M_R)^2}{2\,wl^2}$	5·04	·395	0

The corrected reactions are—

$$R_1 = 347 + 383 + \frac{489 - 283}{5\cdot5} = 768 \text{ lbs.}$$

$$R_2 = 383 + 315 - 38 + \frac{250 - 204}{4\cdot25} = 671 \text{ lbs.}$$

$$R_3 = 315 + 282 - 11 = \underline{586 \text{ lbs.}}$$

$$\underline{\underline{2025 \text{ lbs.}}}$$

BENDING-MOMENTS IN BAYS

Continuing the application of the method to the finding of the magnitudes of the bending-moments at various points in the several bays, we have—

Outer Bay.

$$M_{mid} = \frac{17400}{15960}(-386 \times 1\cdot0215 + 538) \text{ from (2) Appendix}$$

$$= \frac{17400}{15960} \times 143$$

$$= \underline{156 \text{ lbs. ft.}}$$

$$M_{max} = 156 + 5 = \underline{161 \text{ lbs. ft.}} \quad \text{From (3) Appendix.}$$

$$a = 2\cdot75 + \frac{206}{139\cdot2 \times 5\cdot5} = 2\cdot75 + \cdot269 = 3\cdot019 \text{ ft.}$$

From (4)

$$c_1 = 2\cdot75 + \cdot9 \times \cdot269 - 1\cdot2\sqrt{\frac{161}{139\cdot2}} \quad \text{From } (5a)$$

$$= 2\cdot75 + \cdot242 - 1\cdot292 = 1\cdot7 \text{ ft.}$$

$$c_2 = 2\cdot75 + \cdot242 + 1\cdot292 = 4\cdot284 \text{ ft.} \quad \text{From } (5b)$$

Now, using the same formula, we have the following equations for the Intermediate Bay—

Intermediate Bay.

$$M_{mid} = \frac{29200}{23800}(-227 \times 1\cdot04815 + 342)$$

$$= \frac{29200}{23800} \times 104 = \underline{127\cdot5 \text{ lbs. ft.}}$$

$$M_{max} = 127\cdot5 + \cdot395 = \underline{128 \text{ lbs. ft.}}$$

$$a = 2{\cdot}125 + \frac{46}{148{\cdot}3 \times 4{\cdot}25} = 2{\cdot}125 + {\cdot}073 = 2{\cdot}198 \text{ ft.}$$

$$c_1 = 2{\cdot}125 + {\cdot}9 \times {\cdot}073 - 1{\cdot}2\sqrt{\frac{128}{148{\cdot}3}}$$

$$= 2{\cdot}125 + {\cdot}0657 - 1{\cdot}115 = 1{\cdot}076 \text{ ft.}$$

$$c_2 = 2{\cdot}125 + {\cdot}0657 + 1{\cdot}115 = 3{\cdot}306 \text{ ft.}$$

In similar manner for the Centre Bay.

Centre Bay.

$$M_{max} = M_{mid} = \frac{32900}{26724}(-204 \times 1{\cdot}0488 + 287{\cdot}5)$$

$$= \frac{32900}{26724} \times 73{\cdot}5 = 90{\cdot}3 \text{ lbs. ft.}$$

$$c_1 = 2{\cdot}0 - 1{\cdot}2\sqrt{\frac{90{\cdot}3}{141}}$$

$$= 2{\cdot}0 - {\cdot}96 = 1{\cdot}04 \text{ ft.}$$

$$c_2 = 2{\cdot}0 + {\cdot}96 = 2{\cdot}96 \text{ ft.}$$

BENDING-MOMENT DIAGRAM (Top Front Spar)

The diagram of bending-moments for the whole spar may now be plotted, making use of the results just calculated, inserting the offset bending-moments at the points of support, and it will be seen that in such a diagram the curves drawn through the calculated points are quite regular in shape, which, of course, ought to be the case, and which is to some extent a fair check on the work already done. The points in the overhang bending-moment curve are obtained from the bending-moment curve drawn in the load curve in Fig. 5, for it is not strictly accurate to insert a parabolic curve for that portion of the diagram.

On this diagram (Fig. 17) are shown the points, *a*, *b*, *d*, *e*, etc., which constitute the limits of spindling, being actually 3 ins. on either side of the support where the spar is solid, and it is important to remember that at the points of support 1, 2, 3, a bolt $\frac{5}{16}$ in. diameter is inserted, to which is attached the fitting taking the interplane strut, and the wiring plate taking the lift wire. This diagram of bending-moments

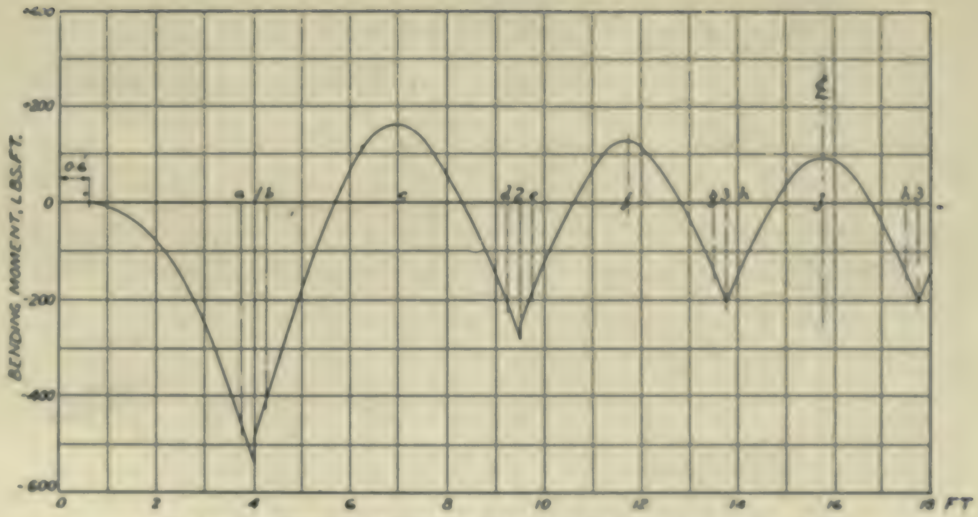


FIG. 17.—BENDING-MOMENT DIAGRAM TOP FRONT SPAR C.P. AT ·3 OF THE CHORD. 5 TIMES LOAD

includes end load bending-moments and is, therefore, far more accurate than that in Fig. 9.

TABLE OF LOADS AND STRESSES

The values of the bending-moments at the several points may now be taken from the diagram and tabulated in some such form as Table 2, which contains also the area and the section modulus for these points as well as further columns necessary to complete the table. It should be noted that in column 5, in which are tabulated the end loads at the various points, these are not the mean end loads as shown in Fig. 11; but represent the exact end loads at the particular points.

The bending-moment stress is obtained by dividing the bending-moment in lbs. ins. by the modulus of the section, whilst the end load stress is equal to the end load divided by the sectional area, and the total stress is the sum of these two stresses. The last column shows the list of factors of safety using a yield stress of 5500 lbs. per square inch, from which it will be observed that the least factor for the whole spar is 1·59.

TABLE 2

TOP FRONT SPAR. FIVE TIMES LOAD, C.P. AT ·3c.

Section.	A in.^{2}	Z in.^{3}	B.M. lbs. ft.	End load. lbs.	B.M. Stress. lbs./in.^{2}	End load Stress. lbs./in.^{2}	Total Stress. lbs./in.^{2}	F,S. on 5500 lbs./in.^{2}
a	2·3	2·32	455	—	2350	—	2350	—
1*a*	4·8	3·17	534	—	2025	—	2025	—
1*b*	4·8	3·17	489	1650	1855	345	2200	—
b	2·3	2·32	405	1650	2100	720	2820	—
c	2·3	2·32	161	1650	835	720	1555	—
d	2·3	2·32	210	1228	1085	535	1620	—
2*d*	4·8	3·17	283	1228	1070	255	1325	—
2*e*	4·8	3·17	250	4826	945	1000	1945	—
e	2·3	2·32	180	4826	930	2100	3030	1·82
f	2·3	2·32	128	5976	665	2600	3265	1·69
g	2·3	2·32	150	5976	775	2600	3375	1·63
3*g*	4·8	3·17	204	5976	770	1245	2015	—
3*h*	4·8	3·17	204	6176	770	1285	2055	—
h	2·3	2·32	150	6176	775	2685	3460	1·59
j	2·3	2·32	90	6176	465	2685	3150	1·75

Shear Stress Due to Bending

The maximum shear stress occurs at the point where the slope of the bending-moment diagram is steepest (*i.e.*, where $\frac{dM}{dx}$ has a maximum value). This point will be seen to occur in the overhang portion, at the point marked 1.

The maximum shear stress is given by the expression—

$$q = \frac{M}{\ }\left\{\left(\frac{B}{B-b} \times \frac{D^2 - d^2)}{8}\right) + \frac{d^2}{8}\right\} \text{ for the hollow section.}$$

$$= \frac{315}{4{\cdot}77}\left\{\left(\frac{1{\cdot}5}{{\cdot}35} \times \frac{4{\cdot}04^2 - 3{\cdot}34^2}{8}\right) + \frac{3{\cdot}34^2}{8}\right\}$$

$$= \frac{315}{4{\cdot}77}\left\{\frac{1{\cdot}5}{{\cdot}35} \times \frac{5{\cdot}15}{8} + 1{\cdot}395\right\} = 275 \text{ lbs. per sq. in.}$$

Allowable shear stress = 800 lbs. per sq. in.

Bottom Front Spar C.P. at ·3 × Chord

By reference to Figs. 6 and 11 it will be observed that this spar is much more lightly loaded than the top front spar, and, furthermore, that the end loads occurring in it are tensile, whereas those on the top front spar are compressive throughout. Generally, this spar is subject to its worst condition of loading when the loading is reversed (*i.e.*, when it sustains down loading and the consequent compressive end loads). Strictly speaking, if the cross-section of the bottom front spar is likely to be similar to that already given for the top front spar, then the stress at any point is bound to be less, since the loading is less, and the end load is tension, both of which help the spar directly. It is usual, however, to make sure that the final reactions do not differ appreciably from those shown in Fig. 11, and to see that the fixing moments are less than the largest on the top front spar. This is done in Table 3 *et seq.*

A final bending-moment diagram for the bottom front spar will not be drawn for this case, and it will be left to the corresponding case for down load to decide finally if the spar section already given is up to strength.

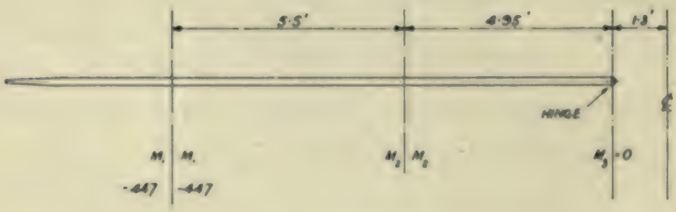


TABLE 3

Span (ft.) . . .	5·5	4·95	
Span (ins.) . . .	66	59·4	
I (in.^{4})	4·77	4·77	
$P_e = \frac{\pi^2 EI}{l^2}$ . . .	17400	21500	$E = 1\cdot6 \times 10^6$
End load P (tension)	174	2650	
$P_e + P$	17574	24150	
w (lbs. per ft.) . .	116·4	122·5	
$\frac{wl^2}{4}$ (l in ft.) . . .	882	752	
$\frac{P}{P_e}$	·0100	·123	
$1 - 0\cdot12\,\frac{P}{P_e}$ . . .	·9988	·98522	
$1 + 0\cdot32\,\frac{P}{P_e}$ . . .	1·0032	1·0394	
$1 + \frac{P}{80\,P_e}$ . . .	1·000125	1·001538	
$\frac{1}{(P_e + P)l}$ . . .	$1\cdot035 \times 10^{-5}$	$\cdot837 \times 10^{-5}$	

The equation for M_2, using the values in Table 3, is (using formula (7) in Appendix)—

$$-447 \times \cdot9988 \times 1\cdot035 \times 10^{-5} + 2M_2 \times 1\cdot0032 \times 1\cdot035 \times 10^{-5} + 882 \times 1\cdot000125 \times 1\cdot035 \times 10^{-5} + 2M_2 \times 1\cdot0394 \times \cdot837 \times 10^{-5} + 752 \times 1\cdot001538 \times \cdot837 \times 10^{-5} = O.$$

$$\therefore\ 2\cdot48M_2 + 2\cdot0788M_2 = -1092 - 753 + 553$$

$$4\cdot559M_2 = -1292 \qquad \therefore\ \underline{M_2 = -283 \text{ lbs. ft.}}$$

The corrected reactions are—

$$R_1 = 289 + 320 + \frac{447 - 283}{5 \cdot 5} = 639 \text{ lbs.}$$

$$R_2 = 320 + 303 - 30 + \frac{283}{4 \cdot 95} = 650 \text{ lbs.}$$

$$R_3 = 303 - 57 = \underline{246 \text{ lbs.}}$$

$$\underline{\underline{1535 \text{ lbs.}}}$$

Corrected Force Diagram

The corrected reactions for the front truss have now been determined, and it will be noticed that they differ but little from the estimated reactions which neglected end load. If there had been any appreciable difference in the final reactions, then it would have been necessary to have drawn a new diagram of loads corresponding to Fig. 11, and to have found the new end loads, etc. In any case, however, if necessary a bending-moment diagram may be drawn, and the actual factors obtained for the bottom front spar as was done for the top

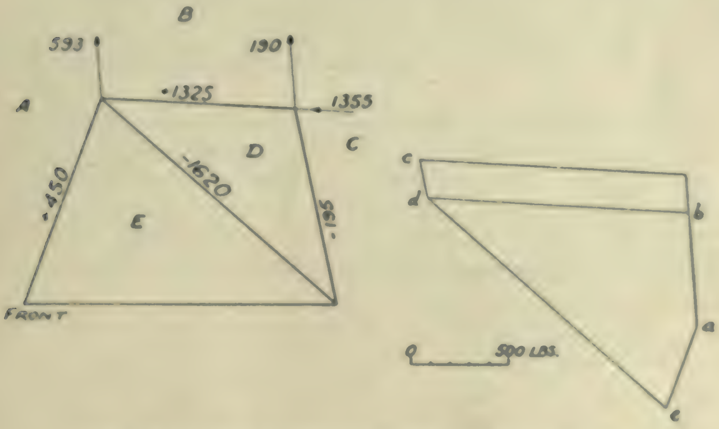


FIG. 18.—DRAG FORCES IN BODY STRUTS, C.P. AT ·3 OF THE CHORD. 5 TIMES LOAD

front spar, but this is hardly needed unless the spar cross section is less in the bottom than the top spar.

True Loads in Lift Wires and Interplane Struts

To estimate the true loads in the lift wires and interplane struts, the loads shown in Fig. 11 must be multiplied by factors obtained from the diagram shown in Fig. 14 (*i.e.*, the loads in the interplane struts and outer lift wires must be increased in the ratio of 1·1 to 1, and the loads in the front main lift wire by 1). This makes allowance for these members being out of the plane of resultant lift.

Drag Forces

The drag forces on the bottom wing and the lift forces at the hinges are taken on to the fuselage direct, but in the case of the top planes the resultant drag force there, together with the local lifts at the body strut joints, are taken on the body struts and the side bracing between them. The actual loads are shown in Fig. 18, the resultant drag being obtained from Fig. 11.

To all intents and purposes this completes the calculations for the case of centre of pressure at ·3 of the chord, and we must next turn our attention to another case, which we have termed case 2, where the centre of pressure is situate at ·5 of the chord. Much of work is similar in character to that in case 1, but it is described fairly fully for the reader's benefit.

CHAPTER IV

CENTRE OF PRESSURE AT ·5 OF THE CHORD

This is the limiting rear position of the centre of pressure, which occurs in ordinary flight, and is the condition which generally obtains for normal horizontal flight with a small angle of incidence of about 3 degrees. The general load curve used is exactly the same as for the centre of pressure forward condition, but the distribution of load on the front and rear spars differs, inasmuch as more load is now taken on the rear spar than the front. The linear scale of the load curve is exactly the same as before, *i.e.*—

$$2{\cdot}5 \text{ ins. represents 1 chord} = 5{\cdot}75 \text{ ft.}$$

$$\therefore \quad 1 \text{ in. represents } \frac{5{\cdot}75}{2{\cdot}5} = 2{\cdot}3 \text{ ft.}$$

Load Diagram For Top Wing

The load diagram must now be set out indicating the positions of the interplane struts, and the centre line of the machine, the wing tip coinciding with the end of the diagram. Fig. 4 shows the position of the centre of pressure at all points along the wing, and it will be seen that it is nearer to the rear spar. From this diagram and that in Fig. 3, the proportion of load taken on the rear spar can be estimated. There is one important feature in which this load curve differs from that for the front spars, for it will be remembered that, in that case, it was assumed that where portions of the trailing edge of the wing are cut away, the centre of pressure moves forward so as to be always at ·3 of the chord, even where the chord is reduced, thus giving the worst condition of loading for the front spar; whereas in this case of C.P. rear, the centre of pressure is assumed to remain at a constant distance from the leading edge, this being the worst condition for the rear spar. The lower portion of the load curve (Fig. 19) represents the proportion of load taken on the rear spar, and the upper portion that taken on the front spar.

Top Rear Spar

As before, the factor to be adopted in this case of C.P. rear

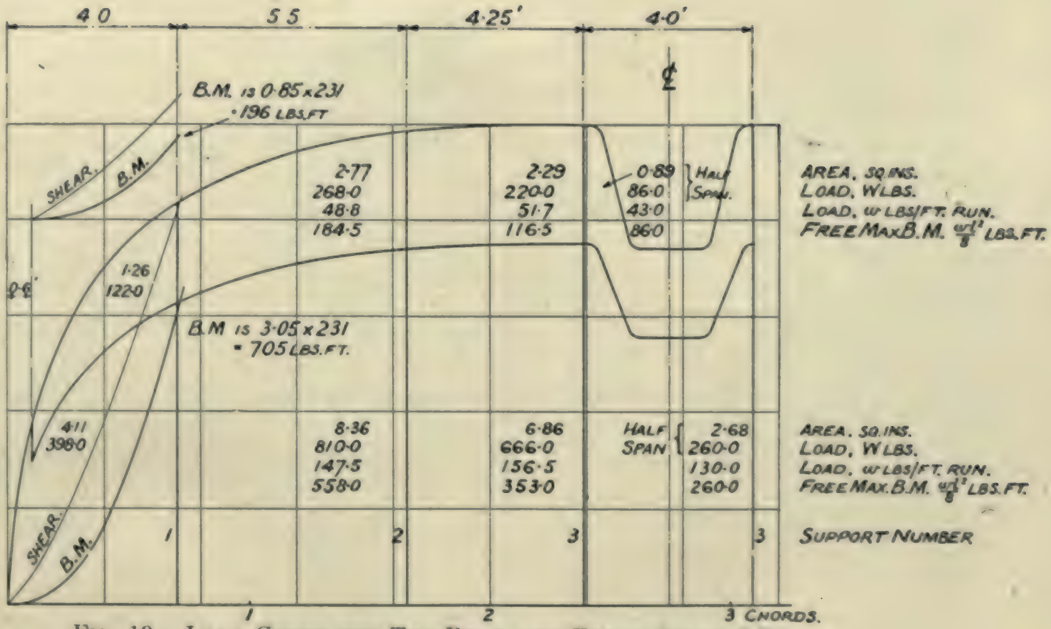


FIG. 19.—LOAD CURVE FOR TOP REAR AND FRONT SPARS C.P AT ·5 OF THE CHORD. 5 TIMES LOAD

is five times load, then the total load on half the top wing = factor × area × load per square foot (net) $= 5 \times \frac{172 \cdot 5}{2} \times 6 \cdot 57 = 2830$ lbs., and the area of the whole load curve representing this load = 29·22 sq. ins., hence the load represented by 1 sq. in. of load curve = 97 lbs.

The area of the diagram representing the load on the spar = 22·01 sq. ins., hence the load on the spar = 22·01 × 97 = 2134 lbs.

Mean height of the load diagram for this spar is

$$\frac{\text{area}}{\text{length}} = \frac{22 \cdot 01}{6 \cdot 85} = 3 \cdot 22 \text{ ins.}$$

then the mean load on the spar

$$= \frac{2134}{15 \cdot 15} = 140 \cdot 8 \text{ lbs. per ft. run}$$

from which the load scale is, therefore,

$$= \frac{\text{Load per foot run}}{\text{mean height of diagram}}$$

$$= \frac{140 \cdot 8}{3 \cdot 22} = 43 \cdot 7 \text{ lbs. per ft. run.}$$

Overhang Bending-Moment

It will be seen from the load diagram that the double integration has been performed graphically, exactly as was done for the front spar in case 1 (Fig. 5), from which the scale of bending-moment can be determined

$$= \text{Load per foot run} \times (\text{linear scale})^2$$

$$1 \text{ in.} = 43 \cdot 7 \times 2 \cdot 3 \times 2 \cdot 3 = 231 \text{ lbs. ft.}$$

hence the maximum bending-moment at the support 1 is—

$$3 \cdot 05 \times 231 = 705 \text{ lbs. ft.}$$

Free Reactions

In order to estimate the offset bending-moments, it is necessary to get some idea of the loads in the outer and inner lift wires, and to do this the free reactions must first be obtained from the completed load diagram.

The load diagram is completed by calculating the areas under the curve between the supports 1–2 and 2–3, and inserting the loads thereon in proportion to the area—thus the load on the overhang = 398 lbs., in the outer bay = 810 lbs. and in the inner bay 666 lbs., whilst the centre portion takes 2 × 260 lbs.

The load per foot is easily determined in each bay, which enables the free bending-moment at the centre of each bay to be calculated.

Bottom Rear Spar

In exactly the same way that the top wing has been dealt with, so a load curve must be drawn for the bottom wing in order to estimate the loading along the bottom spars and the overhang bending-moments.

It is thought to be unnecessary to go into great detail in the text, but simply to make reference to Fig. 20, from which, by comparison with the work already done, all the required information is forthcoming. The same assumption is made regarding the constant distance of the C.P. from the leading edge.

The load on one bottom plane (half wing) is,

$$5{\cdot}43 \times \frac{160}{2} \times 5 = 2172 \text{ lbs.},$$

and the load per square inch of the load curve

$$= \frac{2172}{\text{area under curve}} = \frac{2172}{26{\cdot}8} = 81 \text{ lbs.}$$

Area of the curve representing the load on the bottom rear spar = $20{\cdot}2$ sq. ins., hence the load on the spar

$$= 20{\cdot}2 \times 81 = 1636 \text{ lbs.}$$

The mean height of this diagram

$$= \frac{\text{area}}{\text{length}} = \frac{20{\cdot}2}{6{\cdot}28} = 3{\cdot}22,$$

then mean load per foot run is,

$$\frac{1636}{14{\cdot}45} = 113{\cdot}1 \text{ lbs. per ft. run.}$$

Thus, the load scale

$$= \frac{113{\cdot}1}{3{\cdot}22} = 35{\cdot}2 \text{ lbs. per ft. run.}$$

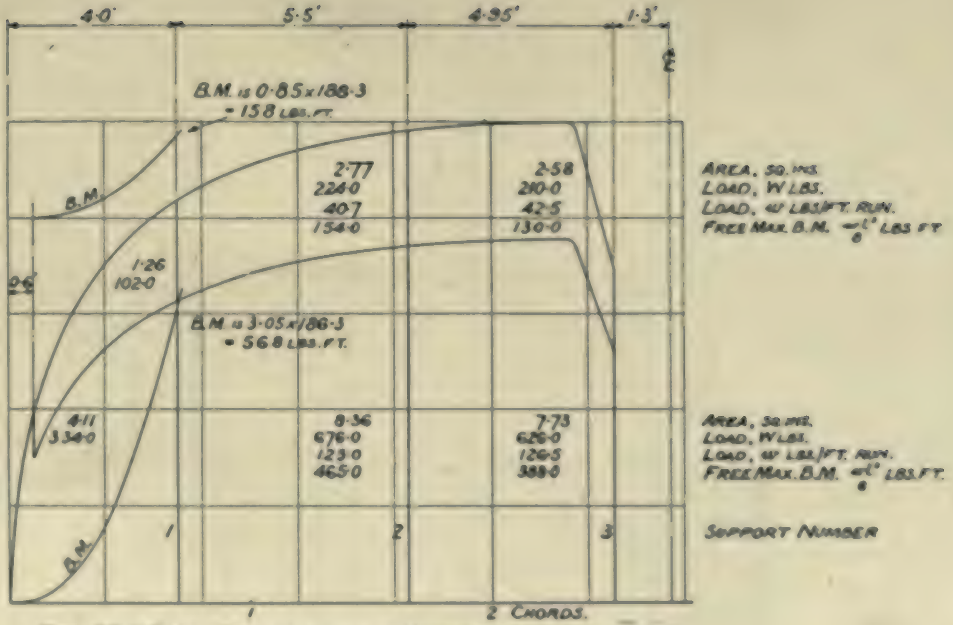


FIG. 20.—LOAD DIAGRAM FOR BOTTOM REAR AND FRONT SPARS C.P. AT ·5 OF THE CHORD. 5 TIMES LOAD

Overhang Bending-Moment

By integrating the load curve twice the overhang bending-moment for the bottom rear spar is determined, the scale of bending-moment being—

$$1 \text{ in.} = 35{\cdot}2 \times 2{\cdot}3 \times 2{\cdot}3 = 186{\cdot}3 \text{ lbs. ft.}$$

Its maximum value occurs at the outer support, and

$$= 186{\cdot}3 \times 3{\cdot}05 = 568 \text{ lbs. ft.}$$

Offset Bending-Moments

The offset bending-moment at the outer and inner points of support can be obtained when the approximate loads in the lift wires have been found, and these in turn depend directly upon the reactions due to lift loads.

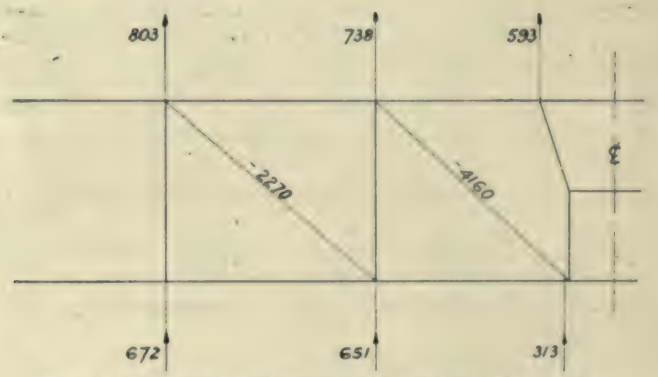


Fig. 21.—Free Reactions Diagram, C.P. at ·5 of the Chord. 5 Times Load

It is easy to approximate to loads in the lift wires by setting out the rear truss, and inserting the free reactions (Fig. 21), for these latter are easily obtained from the load curve as before.

The offset bending-moments at points 1 and 2 on the top rear spar are respectively—

$$2270 \times \frac{\cdot 25}{12} = 47 \text{ lbs. ft., say, } 50 \text{ lbs. ft. at } 1,$$

and

$$4150 \times \frac{\cdot 1}{12} = 34{\cdot}6 \text{ lbs. ft., say, } 35 \text{ lbs. ft. at } 2.$$

Top Front Spar C.P. Rear

By reference to the load curve for the top wing (Fig. 19), the upper portion of which refers to the front spar, it will be seen that a similar procedure has been adopted in order to get at the distribution of the load along the spar, and to find the bending-moment at the overhang, and the free bending-moments in the bays.

The free reactions at the outer and inner points 1 and 2 on the front spar are 256 lbs. and 244 lbs. respectively, and the maximum value of the overhang bending-moment is 231 × ·85 = 196 lbs. ft. By comparison with the free reactions, diagram Fig. 20, the offset bending-moments at points 1 and 2 are 20 lbs. ft. and 15 lbs. ft. respectively, and the offsets are such as to make the offset bending-moments positive in the inner and outer bays.

Bottom Front Spar C.P. Rear

In the same way, by reference to the upper portion of the load curve for the bottom wing, and by similar means, the reactions at the joints 1 and 2, the overhang bending-moment, and the free bending-moments in the bays, may all be determined.

The free reactions at 1 and 2 are found to be 214 lbs. and 217 lbs. respectively, and the maximum value of the overhang bending-moment is 186·3 × ·85 = 158 lbs. ft., by inspection of Fig. 20.

Preliminary Reactions Top Rear Spar

These can be calculated by solving the three moment equations, but it should still be remembered that end load in the spar is not yet accounted for; consequently, the reactions so obtained will be approximate. It should also be noted that the fixing moments obtained in these preliminary estimates are, in every case, those on the *outer* side of the supports.

For the top rear spar, the three-moment equations are—

$$-(705+50)\,5{\cdot}5+2M_2\times5{\cdot}5+2M_2\times4{\cdot}25+4{\cdot}25\,M_3$$
$$=-2\times35\times4{\cdot}25-2\times5{\cdot}5\times558-2\times4{\cdot}25\times353$$

from which—

$$19{\cdot}5\,M_2+4{\cdot}25\,M_3=-5835 \quad . \quad . \quad . \quad (1)$$

Again,

$$4{\cdot}25\,M_2 + 2(4{\cdot}25 + 4)\,M_3 + 4\,M_3 = -\,2 \times 4{\cdot}25 \times 353 - 2 \times 4 \times 260 - 4{\cdot}25 \times 35$$

from which—

$$4{\cdot}25\,M_2 + 20{\cdot}5\,M_3 = -\,5230 \qquad . \quad . \quad . \qquad (2)$$

Hence—

$$M_2 = -\,253 \text{ lbs. ft. and } M_3 = -\,207 \text{ lbs. ft.}$$

The modified reactions are, therefore—

$$\begin{aligned} R_1 &= 398 + 405 + \frac{655 - 253}{5{\cdot}5} &&= 877 \text{ lbs.} \\ R_2 &= 405 + 333 - 74 + \frac{215 - 203}{4{\cdot}25} &&= 667 \text{ lbs.} \\ R_3 &= 333 + 260 - 3 &&= \underline{590 \text{ lbs.}} \\ & &&\quad \underline{2134 \text{ lbs.}} \end{aligned}$$

Top Front Spar C.P. Rear

The top front spar modified reactions may be calculated in a similar manner. The three-moment equations are—

$$-\,(196 + 20)\,5{\cdot}5 + 2\,M_2 \times 5{\cdot}5 + 2\,M_2 \times 4{\cdot}25 + 4{\cdot}25\,M_3 = -\,2 \times 5{\cdot}5 \times 184{\cdot}5 - 2 \times 4{\cdot}25 \times 116{\cdot}5 - 2 \times 15 \times 4{\cdot}25$$

from which—

$$19{\cdot}5\,M_2 + 4{\cdot}25\,M_3 = -\,2180 \qquad . \quad . \quad . \qquad (1)$$

$$4{\cdot}25\,M_2 + 2 \times 8{\cdot}25\,M_3 + 4\,M_3 = -\,2 \times 4{\cdot}25 \times 116{\cdot}5 - 2 \times 4 \times 86 - 4{\cdot}25 \times 15$$

From which—

$$4{\cdot}25\,M_2 + 20{\cdot}5\,M_3 = -\,1742 \qquad . \quad . \quad . \qquad (2)$$

Hence—

$$M_2 = -\,98 \text{ lbs. ft. and } M_3 = -\,65 \text{ lbs. ft.,}$$

and the modified reactions are—

$$\begin{aligned} R_1 &= 122 + 134 + \frac{176 - 98}{5{\cdot}5} &&= 270 \text{ lbs.} \\ R_2 &= 134 + 110 - 4 + \frac{83 - 65}{4{\cdot}25} &&= 234 \text{ lbs.} \\ R_3 &= 110 + \ 86 - 4 &&= \underline{192 \text{ lbs.}} \\ & &&\quad \underline{696 \text{ lbs.}} \end{aligned}$$

Bottom Rear Spar

It will be noted that the bottom wing is hinged, and that there are no offsets to contend with, hence the three-moment equation for the bottom rear spar is—

$$-568 \times 5{\cdot}5 + 2 \times (5{\cdot}5 + 4{\cdot}95)\, M_2 + 4{\cdot}95 \times O = -2 \times 5{\cdot}5 \times 465 - 2 \times 4{\cdot}95 \times 388$$

Since $M_3 = O$ because of the hinge, this becomes—

$$20{\cdot}9\, M_2 = -5110 - 3840 + 3130 = -5820,$$

from which—

$$M_2 = -279 \text{ lbs. ft.}$$

The reactions are, therefore—

$$R_1 = 334 + 338 + \frac{568 - 279}{5{\cdot}5} = 725 \text{ lbs.}$$

$$R_2 = 338 + 313 - 53 + \frac{279}{4{\cdot}95} = 654 \text{ lbs.}$$

$$R_3 = 313 - 56 = 257 \text{ lbs.}$$

$$\underline{1636 \text{ lbs.}}$$

Bottom Front Spar C.P. Rear

Similarly, for bottom front spar, the three-moment equation is—

$$-158 \times 5{\cdot}5 + 2\,(5{\cdot}5 + 4{\cdot}95)\, M_2 + O = -2 \times 5{\cdot}5 \times 154 - 2 \times 4{\cdot}95 \times 130.$$

Thus—

$$20{\cdot}9\, M_2 = -1695 - 1288 + 870 = -2113$$

From which—

$$M_2 = -101 \text{ lbs. ft.}$$

The reactions are—

$$R_1 = 102 + 112 + \frac{158 - 101}{5{\cdot}5} = 224 \text{ lbs.}$$

$$R_2 = 112 + 105 - 10 + \frac{101}{4{\cdot}95} = 227 \text{ lbs.}$$

$$R_3 = 105 - 20 = 85 \text{ lbs.}$$

$$\underline{536 \text{ lbs.}}$$

PRELIMINARY BENDING-MOMENT

Diagrams for Top and Bottom Rear Spars

At this point it must be our endeavour to construct for the top and bottom rear spars the preliminary bending-moment

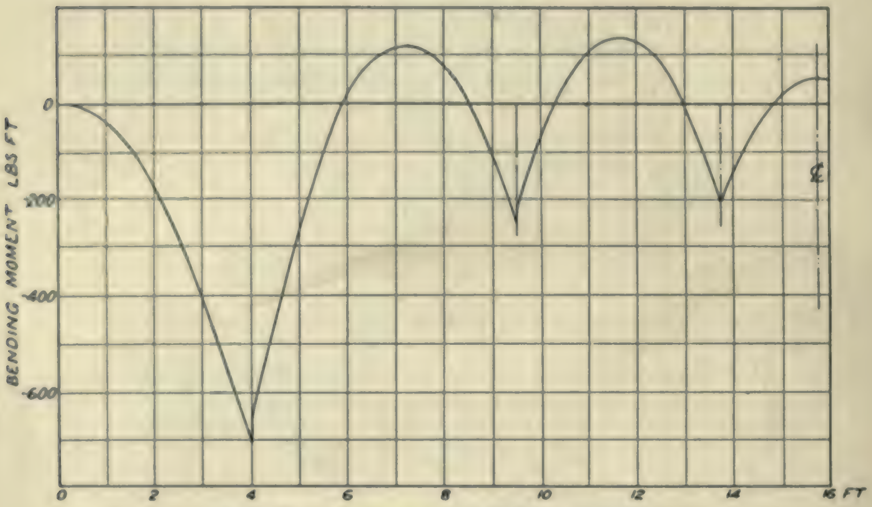


FIG. 22.—PRELIMINARY B.M. DIAGRAM.
TOP REAR SPAR

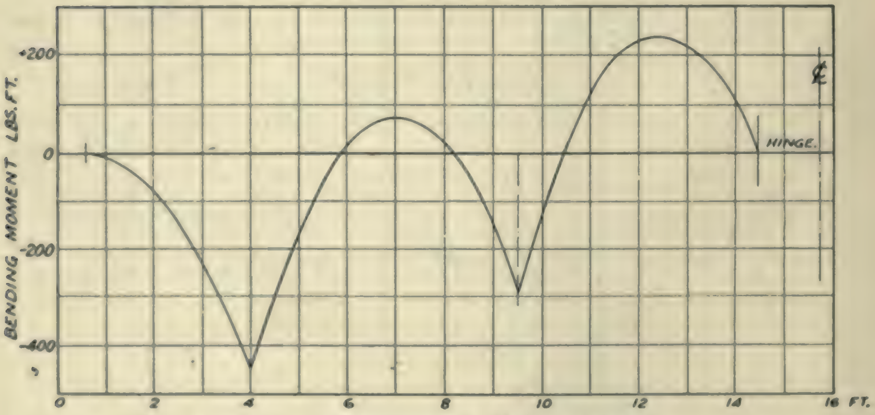


FIG. 23.—PRELIMINARY B.M. DIAGRAM.
BOTTOM REAR SPAR

diagrams which will enable us to make a good estimate of the general overall sizes of the spar sections.

Set out first of all the points of support, and the bending-moments at these points, which have already been found, the overhang B.M. being obtained from the load diagrams. (Figs. 19 and 20.)

Insert now the offset bending-moments which are already known, and also the parabolic portions of the diagram due to uniform air lift forces, whose maximum ordinates may be obtained by inspection from the load curve.

From the experience gained in constructing similar diagrams for the front top and bottom spars, there will be no difficulty in completing both diagrams as shown in Figs. 22 and 23.

The end loads must now be determined, and to do this a diagram of struts and wires must be made.

DIAGRAM OF STRUTS AND WIRES

This diagram (Fig. 24) may now be drawn, making use of the approximate reactions just found, inserting the front truss at the bottom of the sheet, since, for the case of C.P. rear, the front truss is only of secondary importance, and is mainly used to obtain correct drag forces. It will not be needful to go again into details describing how to obtain the end loads in the spars and the loads in the wires, but if any difficulty ever presents itself, the work can always be checked graphically. The drag forces will be found as follows—

DRAG FORCES

The attitude of the aeroplane, when flying horizontally with C.P. at ·5 of the chord, is such that the angle of incidence of the wings is about 3 degrees, and it is advisable to draw an end view of the wing structure showing the chord set at 3 degrees to the horizontal, together with the struts and wires in their relative positions. The drag forces on the wings may now be found, either by drawing out a separate diagram for each joint, as was done in case (1), or by the second method, in which only one diagram is made.

Fig. 25 is the diagram corresponding to Fig. 14 for C.P. forward. As before, the lift force is set perpendicularly to the line of flight, and the direction of resultant lift obtained by

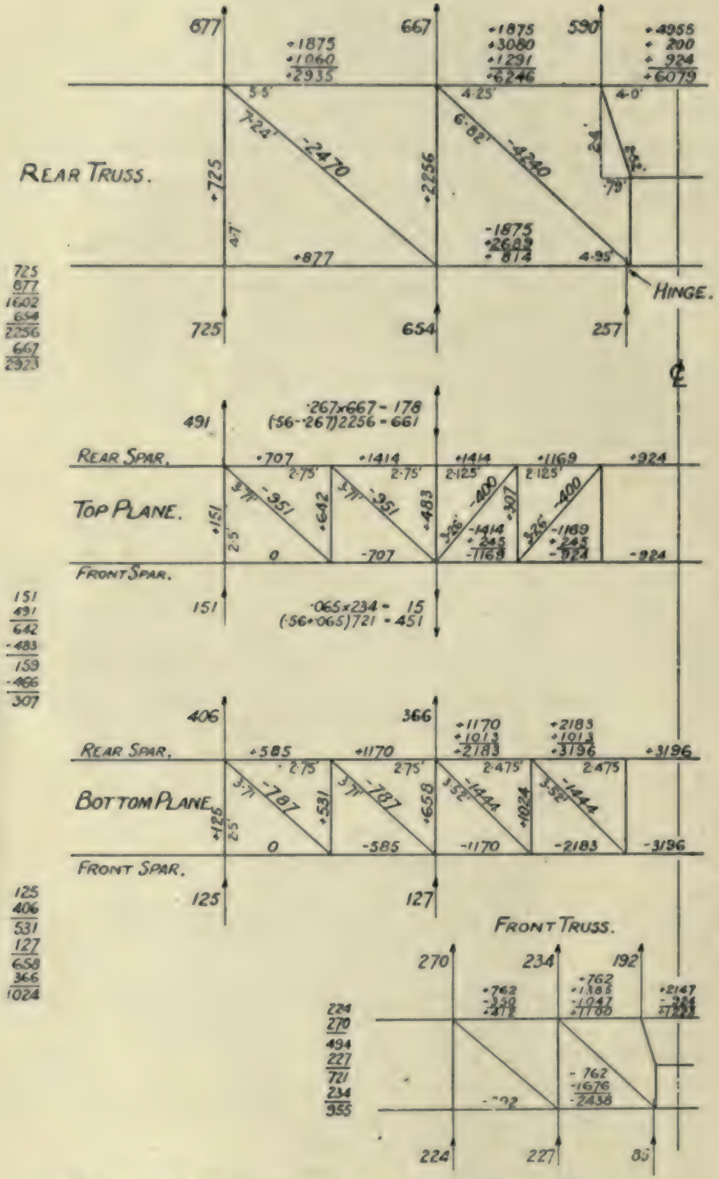


FIG. 24.—FORCE DIAGRAM FOR REAR TRUSS, C.P. AT ·5 OF THE CHORD. 5 TIMES LOAD

taking the drag equal to one-seventh of the lift. This resultant lift is marked off unit length, *OA*, to some scale, and a line drawn parallel to wing chord (*i.e.*, 3 degrees to horizontal), whilst through the point *O* the lines *OB*, *OC*, *OD* are drawn parallel to the struts and lift wires to cut the line through *A*, in *B*, *C*, and *D*. The drag force at each joint may now be calculated, being dependent upon the lift force at each joint in exactly the same way as described in detail for case (1), C.P. forward. The values obtained are inserted in Fig. 24, and the end loads in the spars and loads in the drag bracing on both the top and bottom wings calculated or obtained by graphical methods.

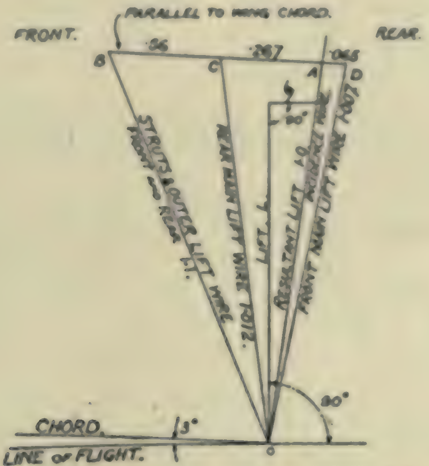


FIG. 25.—DRAG COEFFICIENT DIAGRAM, C.P. AT ·5 OF THE CHORD. ANGLE OF INCIDENCE 3 DEGREES

PRELIMINARY SECTION OF REAR SPARS

Top Rear Spar

At this point we are able, by taking account of the B.M. diagram and the end loads for the spar, to get some idea of the probable size of section suitable to the loading, and so a trial section is to be assumed. The depth of section is fixed by the aerofoil section, and the depth will be found to be about 3·3 in.

Assuming an overall width of 1·5 in. and web and flange thicknesses of ·4 in., it will be found that the section modulus is given by—

$$Z = \frac{BD^3 - bd^3}{6D} = \frac{1{\cdot}5 \times 3{\cdot}3^3 - 1{\cdot}1 \times 2{\cdot}5^3}{6 \times 3{\cdot}3}$$

$$= \frac{54 - 17{\cdot}2}{19{\cdot}8} = 1{\cdot}86 \text{ in.}^3$$

Assuming section to be split into rectangles,

$$\text{Area of section} = 3{\cdot}3 \times 1{\cdot}5 - 1{\cdot}1 \times 2{\cdot}5$$
$$= 4{\cdot}95 - 2{\cdot}75 = 2{\cdot}2 \text{ in.}^2$$

Judging from the B.M. diagram and the end loads it is probable that the stress will be a maximum just inside the outer support.

$$\text{B.M. stress} = \frac{653 \times 12}{1{\cdot}86} = 4220 \text{ lbs. per sq. in.}$$

$$\text{End load stress} = \frac{2582}{2{\cdot}2} = 1170 \text{ lbs. per sq. in.}$$

$$\text{Total stress} = 5390 \text{ lbs. per sq. in.}$$

The end load bending-moment has been neglected in this estimate, and on that account the final stress may be slightly greater than the figure now obtained, but against that we have neglected the fact that the spar is solid for a distance of 3 in. from the point of support where the B.M. was taken which would mean that when account is taken of solid portion the stress will be reduced.

Possibly these two factors more or less counterbalance, and it is quite possible that the final stress will be just below 5500 lbs. per square inch, in which case the trial section assumed may be considered up to strength.

Bottom Rear Spar

Following similar lines a preliminary bending-moment diagram for the rear bottom spar may be constructed, since all the necessary particulars are now available.

This preliminary B.M. diagram will be seen from Fig. 23, and assuming that the bottom rear spar has the same section as the top rear spar it will be found that the maximum stress is within the safe limit. This maximum point will be found to be just inside the outer support when the B.M. stress

$$= \frac{\text{B.M.}}{\text{modulus of section}} = \frac{568 \times 12}{1{\cdot}86}$$

$$= 3670 \text{ lbs. per sq. in.}$$

$$\text{End load stress} = \frac{\text{end load}}{\text{sectional area}} = \frac{877}{2{\cdot}2}$$

$$= 400 \text{ lbs. per sq. in.}$$

$$\text{Total stress} = 4070 \text{ lbs. per sq. in.}$$

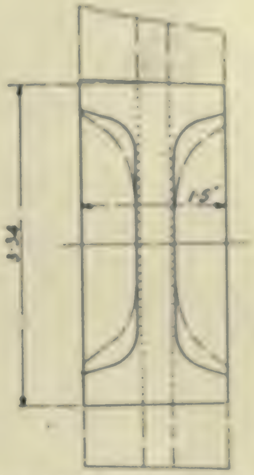


FIG. 26.—M.I. DIAGRAM. REAR SPARS

FINAL ESTIMATE OF SPAR STRESSES

We are now in a position to make a final estimate of the stresses in the spars induced by this case of centre of pressure at ·5 of the chord. The exact moment of inertia, section modulus, and area of section ought first to be calculated from the correct section shown in Fig. 26, where the cube figure is also shown.

Herewith are the results required for various sections of the spar—

REAR SPARS

$I = \frac{A}{3} \times 2$, since the cube scale is $\frac{1}{2}$ full size.

	Area of Cube Figure.	I	Z	Area of Section.
Solid section . . .	6·98 sq. ins.	4·65 ins.^{4}	2·78 ins^3.	5·01 sq. ins.
Solid section with $\frac{5}{16}$ ins. hole	5·54 ,,	3·69 ,,	2·21 ,,	3·97 ,,
Hollow section . . .	4·75 ,,	3·17 ,,	1·9 ,,	2·17 ,,

Top Rear Spar

Applying now the more exact method described in the appendix to these pages, the tabulation in Table 4 shows the various quantities required, and the general procedure may be better followed by reference to it and the work which follows immediately after it.

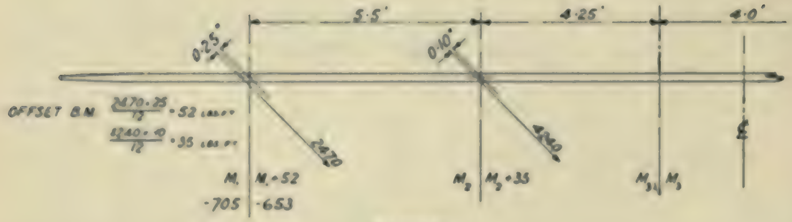

5·5'
4·25'
4·0'
0·25"
0·10"
OFFSET B.M. $\frac{2470 \times 25}{12}$ = 52 lbs ft
$\frac{1240 \times 10}{12}$ = 35 lbs ft
2470
4240
℄
M_1
$M_1 + 52$
–705
–653
M_2
$M_2 + 35$
M_{3_L}
M_3

TABLE 4

Span (ft.)	5·5	4·25	4·0
Span (ins.)	66	51	48
I (in.^{4})	3·17	3·17	3·17
$P_e = \frac{\pi^2 E I}{l^2}$	11550	19350	21900
End load P (Comp.)	2935	6246	6079
$P_e - P$	8615	13104	15821
w (lbs. per ft.)	147·5	156·5	130·0
$\frac{wl^2}{4}$	1116	706	520
$\frac{P}{P_e}$	0·254	0·3225	0·2775
$1 + 0{\cdot}2\frac{P}{P_e}$	1·0508	1·0645	1·0555
$1 - 0{\cdot}38\frac{P}{P_e}$	0·9035	0·8775	0·8945
$1 - \frac{P}{70 P_e}$	0·99637	0·99539	0·99603
$\frac{1}{(P_e - P)l}$	$2{\cdot}11 \times 10^{-5}$	$1{\cdot}795 \times 10^{-5}$	$1{\cdot}58 \times 10^{-5}$
M_R M_L	–653 –267	–232 –214	–214 –214
$\frac{1}{2}(M_R + M_L)$	–460	–223	–214
$1 + {\cdot}26\frac{P}{P_e}$	1·0660	1·0839	1·0722
$1{\cdot}02\left(\frac{wl^2}{8}\right)$	569	360	265
$(M_L - M_R)$	386	18	0
$\frac{(M_L - M_R)^2}{2\,wl^2}$	16·65	0·0575	0

FINAL REACTIONS

Top Rear Spar

Having completed Table 4, formula (1) in Appendix is applied to find the fixing moments M_{2_L} and M_{3_L} at points 2 and 3, as follows—

$$- 653 \times 1{\cdot}0508 \times 2{\cdot}11 \times 10^{-5} + 2\,M_{2_L} \times {\cdot}9035 \times 2{\cdot}11 \times 10^{-5} + 1116 \times {\cdot}9964 \times 2{\cdot}11 \times 10^{-5} + M_{3_L} \times 1{\cdot}0645 \times 1{\cdot}795 \times 10^{-5} + 2(M_{2_L} + 35)\,{\cdot}8775 \times 1{\cdot}795 \times 10^{-5} + 706 \times {\cdot}9954 \times 1{\cdot}795 \times 10^{-5} = O.$$

Dividing through by $1{\cdot}795 \times 10^{-5}$ we have

$$-807{\cdot}5 + 2{\cdot}12\,M_{2_L} + 1{\cdot}755\,M_{2_L} + 1{\cdot}0645\,M_{3_L} = -1307 - 703 - 61{\cdot}5$$

$$\therefore\ 3{\cdot}875\,M_{2_L} + 1{\cdot}0645\,M_{3_L} = -1264{\cdot}0 \qquad . \qquad . \qquad (A)$$

Again, for the two inner bays, top rear spar, we have

$$(M_{2_L} + 35)\,1{\cdot}0645 \times 1{\cdot}795 \times 10^{-5} + 2\,M_{3_L} \times {\cdot}8775 \times 1{\cdot}795 \times 10^{-5} + 706 \times {\cdot}9954 \times 1{\cdot}795 \times 10^{-5} + M_{3_L} \times 1{\cdot}0555 \times 1{\cdot}58 \times 10^{-5} + 2\,M_{3_L} \times {\cdot}8945 \times 1{\cdot}58 \times 10^{-5} + 520 \times {\cdot}996 \times 1{\cdot}58 \times 10^{-5} = O.$$

Dividing through by $1{\cdot}58 \times 10^{-5}$ we have

$$1{\cdot}21\,M_{2_L} + 42{\cdot}3 + 1{\cdot}992\,M_{3_L} + 1{\cdot}0555\,M_{3_L} + 1{\cdot}789\,M_{3_L} = -797 - 518$$

$$\therefore\ 1{\cdot}21\,M_{2_L} + 4{\cdot}836\,M_{3_L} = -1357{\cdot}3 \qquad . \qquad . \qquad . \qquad (B)$$

Dividing (A) by 3·205,

$$\begin{array}{r}
1{\cdot}21\,M_{2_L} + 0{\cdot}3325\,M_{3_L} = -395{\cdot} \\
1{\cdot}21\,M_{2_L} + 4{\cdot}836\,M_{3_L} = -1357{\cdot}3 \\
\hline
4{\cdot}5035\,M_{3_L} = -\ \ 962{\cdot}3 \\
\underline{M_{3_L} = -214 \text{ lbs. ft.}}
\end{array}$$

$$\text{From (A)}\ M_{2_L} = \frac{-1264 + 228}{3{\cdot}875}$$

$$= -267 \text{ lbs. ft.}$$

$$R_1 = 398 + 405 + \frac{653 - 267}{5{\cdot}5} = 873 \text{ lbs.}$$

$$R_2 = 405 + 333 - 70 + \frac{232 - 214}{4{\cdot}25} = 672 \text{ lbs.}$$

$$R_3 = 333 + 260 - 4 = 589 \text{ lbs.}$$

$$\underline{2134 \text{ lbs.}}$$

Note that these corrected reactions are practically the same as those preliminary estimated; hence the diagram of struts and wires (Fig. 24) requires no modification.

BENDING-MOMENTS IN BAYS

Using the quantities from Table 4 and applying the formulae (2, 3, 4, 5*a*, 5*b* in Appendix) to find the bending-moments at various points in the different bays, we have—

Outer bay—

$$M_{mid} = \frac{11550}{8615}(-460 \times 1{\cdot}066 + 569)$$

$$= \frac{11550}{8615} \times 78 = \underline{104{\cdot}5\text{ lbs. ft.}} \qquad \text{From (2)}$$

$$M_{max} = 104{\cdot}5 + 16{\cdot}65 \qquad = \underline{121{\cdot}15\text{ lbs. ft.}} \quad \text{From (3)}$$

$$a = 2{\cdot}75 + \frac{386}{147{\cdot}5 \times 5{\cdot}5} = 2{\cdot}75 + {\cdot}476 = \underline{3{\cdot}226\text{ ft.}}$$

From (4)

$$c_1 = 2{\cdot}75 + {\cdot}9 \times {\cdot}476 - 1{\cdot}2\sqrt{\frac{121{\cdot}15}{147{\cdot}5}} \qquad \text{From } (5a)$$

$$= 2{\cdot}75 + {\cdot}429 - 1{\cdot}086$$

$$= \underline{2{\cdot}093\text{ ft.}}$$

$$c_2 = 2{\cdot}75 + {\cdot}429 + 1{\cdot}086 \qquad \text{From } (5b)$$

$$= \underline{4{\cdot}265\text{ ft.}}$$

Intermediate bay (using same formulae)—

$$M_{mid} = \frac{19350}{13104}(-223 \times 1{\cdot}0839 + 360)$$

$$= \frac{19350}{13104} \times 118 \qquad = \underline{174\text{ lbs. ft.}}$$

$$M_{max} = 174 \quad + {\cdot}0575 \qquad = \underline{174\text{ lbs. ft.}}$$

$$a = 2{\cdot}125 + \frac{18}{156{\cdot}5 \times 4{\cdot}25} = 2{\cdot}125 + {\cdot}027$$

$$= \underline{2{\cdot}152 \text{ ft.}}$$

$$c_1 = 2{\cdot}125 + {\cdot}9 \times {\cdot}027 - 1{\cdot}2\sqrt{\frac{174}{156{\cdot}5}}$$

$$= 2{\cdot}125 + {\cdot}024 - 1{\cdot}265 = \underline{0{\cdot}884 \text{ ft.}}$$

$$c_2 = 2{\cdot}125 + {\cdot}024 + 1{\cdot}265 = \underline{3{\cdot}414 \text{ ft.}}$$

Similarly for Centre Bay.

Centre Bay—

$$M_{mid} = M_{max} = \frac{21900}{15821}(-214 \times 1{\cdot}0722 + 265)$$

$$= \frac{21900}{15821} \times 36 = \underline{50 \text{ lbs. ft.}}$$

$$c_1 = 2{\cdot}0 - 1{\cdot}2\sqrt{\frac{50}{130}}$$

$$= 2{\cdot}0 - {\cdot}745 = \underline{1{\cdot}255 \text{ ft.}}$$

$$c_2 = 2{\cdot}0 + {\cdot}745 = \underline{2{\cdot}745 \text{ ft.}}$$

The diagram of bending-moments may now be plotted from the calculated figures, and this has been done in Fig. 27, where it will be noticed that the offset bending-moments have also been inserted at the points of support, and since the curves are of regular form the diagram constitutes a sort of check on the previous work.

The overhang bending-moment diagram is simply transferred from the load curve (Fig. 19) by altering the scale as

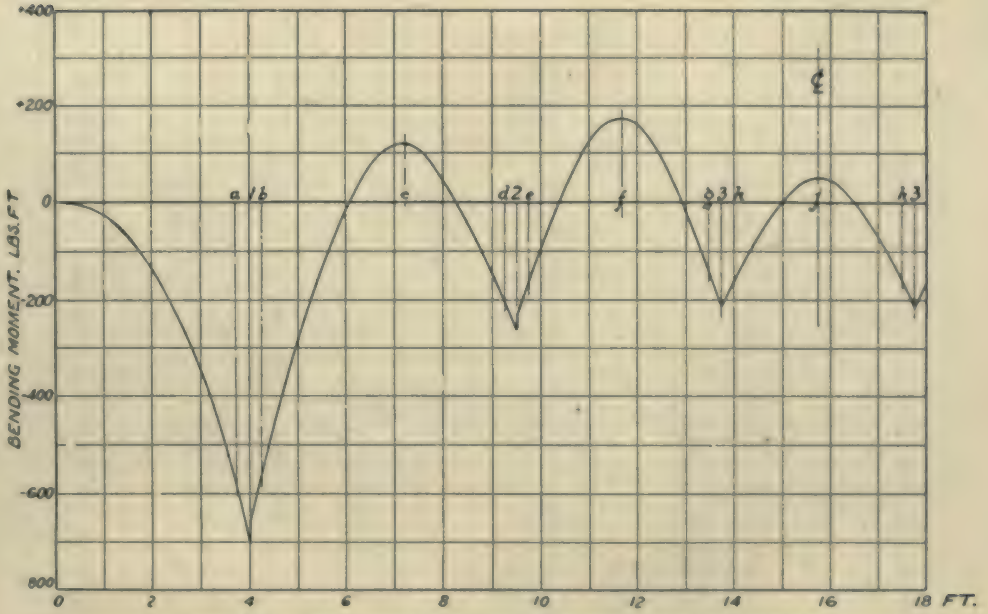


FIG. 27.—BENDING-MOMENT DIAGRAM FOR TOP REAR SPAR, C.P. AT ·5 OF THE CHORD. 5 TIMES LOAD

necessary. The points where spindling begins are clearly shown in each bay on the diagram, and it is usual to estimate the stresses at these points as well as on either side of each support, where a 5-16th in. vertical bolt passes through the spar.

Table of Stresses in Top Rear Spar

Table 5 is a tabulation of the end loads and the bending-moments at the various well-defined points along the spar, which points are chosen such that if the factors of safety are satisfactory there, then in general the whole spar will be up to strength. It should be noted that the end loads given in Table 5 are exact end loads at the particular points, and are not the mean loads shown in Fig. 24.

The least factor obtained is just over five at the point *b*, hence the spar is up to strength.

The calculations for the solid and hollow sections of the top rear spar are tabulated under Fig. 26, the method being exactly the same as was used for the front spars.

TABLE 5

TOP REAR SPAR. FIVE TIMES LOAD, C.P. AT ·5c.

Section.	A in.^{2}	Z in.^{3}	B.M. lbs. ft.	End Load lbs.	B.M. Stress lbs./in.^{2}	E.L. Stress lbs./in.^{2}	Total Stress, lbs./in.^{2}	F.S. on 5500 lbs./sq. in.
a	2·17	1·90	600	—	3800	—	3800	1·45
1*a*	3·97	2·21	705	—	3830	—	3830	1·44
1*b*	3·97	2·21	653	2582	3550	650	4200	1·31
b	2·17	1·90	550	2582	3480	1190	4670	1·19
c	2·17	1·90	121	3289	765	1515	2280	—
d	2·17	1·90	200	3289	1265	1515	2780	—
2*d*	3·97	2·21	267	3289	1450	830	2280	—
2*e*	3·97	2·21	232	6369	1260	1605	2865	—
e	2·17	1·90	165	6369	1045	2940	3985	1·38
f	2·17	1·90	174	6369	1100	2940	4040	1·36
g	2·17	1·90	150	6124	950	2820	3770	1·46
3*g*	3·97	2·21	214	6124	1165	1545	2710	—
3*h*	3·97	2·21	214	6079	1165	1530	2695	—
h	2·17	1·90	160	6079	1010	2800	3810	1·44
j	2·17	1·90	50	6079	320	2800	3120	—

Shear Stress Due to Bending

This is a maximum near the outer support in the overhang portion, where the slope of the bending-moment diagram is a maximum. (*See* Fig. 27.)

The shear stress on a hollow section is given by—

$$q = \frac{F}{I}\left\{\frac{B}{B-b}\left(\frac{D^2-d^2}{8}\right)+\frac{d^2}{8}\right\}$$

where F and I are shearing force and moment of inertia respectively at the point in question, and B, b, D and d are breadths and depths.

The value of the shearing force at that point = 405 lbs., obtained from slope of curve.

$$\therefore\ q = \frac{405}{3\cdot17}\left\{\left(\frac{1\cdot5}{\cdot40}\times\frac{3\cdot34^2-2\cdot54^2}{8}\right)+\frac{2\cdot54^2}{8}\right\}$$

$$= \frac{405}{3\cdot17}\left(\frac{1\cdot5\times4\cdot7}{3\cdot2}+\cdot806\right) = \frac{405\times3\cdot011}{3\cdot17}$$

= 385 lbs. per square inch, which is quite allowable.

BOTTOM REAR SPAR

Final Estimation of Stresses

The bottom rear spar must be dealt with in a similar way (*i.e.*, the final fixing moments must be determined, and the final reactions found and compared with the approximate ones). From Table 6 and the equations immediately following, the final reactions are estimated, and do not differ appreciably from the approximate reactions; hence, Fig. 24 may be treated as a final diagram as well as a preliminary one; thus the end loads undergo no alteration.

The calculations are shown in Table 6.

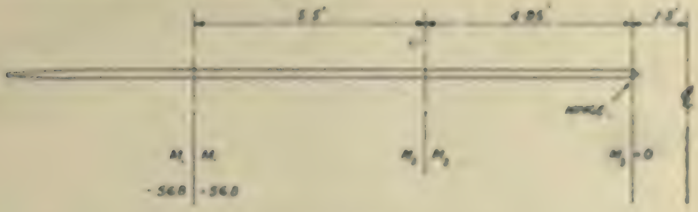


TABLE 6

Span (ft.)	5·5	4·95
Span (ins.)	66·0	59·4
I (in⁴).	3·17	3·17
$P_e = \dfrac{\pi 2EI}{l^2}$	11550	14250
End load P (Comp.)	877	814
$P_e - P$	10673	13436
w (lbs. per ft.).	123·0	126·5
$\dfrac{wl^2}{4}$	930	776
$\dfrac{P}{P_e}$	0·0759	0·0571
$1 + 0{\cdot}2\dfrac{P}{P_e}$	1·01518	1·01142
$1 - 0{\cdot}38\dfrac{P}{P_e}$	0·97115	0·9783
$1 - \dfrac{P}{70P_e}$	0·998915	0·999185
$\dfrac{1}{(P_e - P)l}$	$1{\cdot}705 \times 10^{-5}$	$1{\cdot}502 \times 10^{-5}$
$M_R \quad M_L$	−568 −282	−282 0
$\frac{1}{2}(M_R + M_L)$	−425	−141
$1 + {\cdot}26\dfrac{P}{P_e}$	1·01975	1·01485
$1{\cdot}02\left(\dfrac{wl^2}{8}\right)$	475	395
$(M_L - M_R)$	286	282
$\dfrac{(M_L - M_R)^2}{2wl^2}$	11·0	12·8

Then, applying formula 1 in Appendix—

$$
\begin{aligned}
&-568 \times 1{\cdot}0152 \times 1{\cdot}705 \times 10^{-5} \\
&\qquad\qquad + 2\,M_{2_L} \times {\cdot}9712 \times 1{\cdot}705 \times 10^{-5} \\
&+ 930 \times {\cdot}9989 \times 1{\cdot}705 \times 10^{-5} \\
&\qquad\qquad + 2\,M_{2_L} \times {\cdot}9783 \times 1{\cdot}502 \times 10^{-5} \\
&+ 776 \times {\cdot}9992 \times 1{\cdot}502 \times 10^{-5} = O
\end{aligned}
$$

Dividing through by $1{\cdot}502 \times 10^{-5}$ we obtain

$$2{\cdot}21\, M_{2_L} + 1{\cdot}957\, M_{2_L} = -1055 - 775 + 655$$

$$4{\cdot}167\, M_{2_L} = -1175$$

$$\therefore \quad M_{2_L} = \underline{-282 \text{ lbs. ft.}}$$

$$R_1 = 334 + 338 + \frac{568 - 282}{5{\cdot}5} = 724 \text{ lbs.}$$

$$R_2 = 338 + 313 - 52 + \frac{282}{4{\cdot}95} = 656 \text{ lbs.}$$

$$R_3 = 313 - 57 = 256 \text{ lbs.}$$

$$\underline{1636 \text{ lbs.}}$$

BENDING-MOMENTS IN BAYS

The calculation of bending-moment at several points along the spar are given herewith.

For the B.M. diagram we have—

Outer bay—

$$M_{mid} = \frac{11550}{10673}(-425 \times 1{\cdot}01975 + 475)$$

$$= \frac{11550}{10673} \times 42 = \underline{45{\cdot}5 \text{ lbs. ft.}} \qquad \text{From (2)}$$

$$M_{max} = 45{\cdot}5 + 11 = \underline{56{\cdot}5 \text{ lbs. ft.}} \qquad \text{From (3)}$$

$$a = 2{\cdot}75 + \frac{286}{123 \times 5{\cdot}5} = 2{\cdot}75 + {\cdot}422$$

$$= \underline{3{\cdot}172 \text{ ft.}} \qquad \text{From (4)}$$

$$c_1 = 2{\cdot}75 + {\cdot}9 \times {\cdot}422 - 1{\cdot}2\sqrt{\frac{56{\cdot}5}{123}}$$

$$= 2{\cdot}75 + {\cdot}38 - {\cdot}813 = \underline{2{\cdot}317 \text{ ft.}} \qquad \text{From (5}a\text{)}$$

$$c_2 = 2{\cdot}75 + {\cdot}38 + {\cdot}813 = \underline{3{\cdot}943 \text{ ft.}}$$

Similarly for Inner Bay—

Inner bay—

$$M_{mid} = \frac{14250}{13436}(-141 \times 1{\cdot}01485 + 395)$$

$$= \frac{14250}{13436} \times 252 \qquad = \underline{267 \text{ lbs. ft.}}$$

$$M_{max} = 267 + 12{\cdot}8 \qquad = \underline{279{\cdot}8 \text{ lbs. ft.}}$$

$$a = 2{\cdot}475 + \frac{282}{126{\cdot}5 \times 4{\cdot}95} = 2{\cdot}475 + {\cdot}45$$

$$= \underline{2{\cdot}925 \text{ ft.}}$$

$$c_1 = 2{\cdot}475 + {\cdot}9 \times {\cdot}45 - 1{\cdot}2\sqrt{\frac{279{\cdot}8}{126{\cdot}5}}$$

$$= 2{\cdot}475 + {\cdot}405 - 1{\cdot}785 = \underline{1{\cdot}095 \text{ ft.}}$$

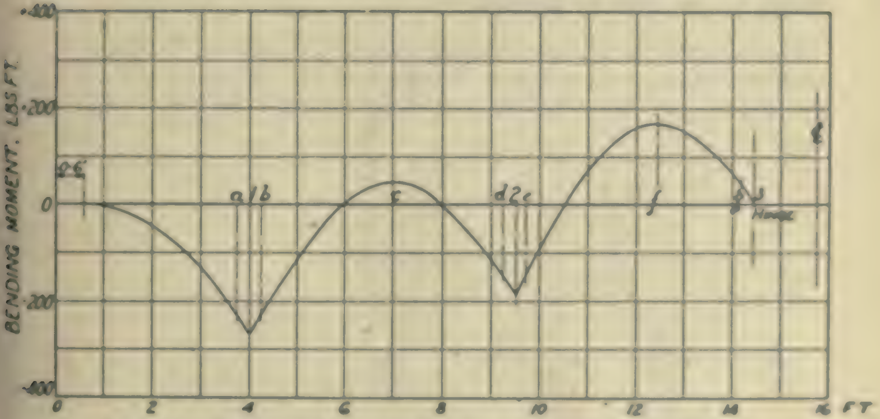


FIG. 28.—BENDING-MOMENT DIAGRAM FOR BOTTOM REAR SPAR C.P. AT ·5 OF THE CHORD. 5 TIMES LOAD

Fig. 28 shows the bending-moment diagram constructed from the particulars just determined for the bottom rear spar, and the points *a*, *b*, *d*, *e*, etc., show the extent of the solid portions. The stresses due to combined end load and bending-moment are set forth in Table 7, where the section is assumed to be identical with that of the top rear spar.

TABLE 7

BOTTOM REAR SPAR. FIVE TIMES LOAD, C.P. AT ·5c.

Section.	A in.²	Z in.³	B.M. lbs. ft.	End Load. lbs.	B.M. Stress, lbs./in.²	E.L. Stress, lbs./in.²	Total Stress, lbs./in.²	F/S on 5500 lbs./sq. in.
a	2·17	1·90	490	—	3100	—	3100	1·77
1a	3·97	2·21	568	—	3090	—	3090	—
1b	3·97	2·21	568	585	3090	150	3240	1·70
b	2·17	1·90	490	585	3100	270	3370	1·63
c	2·17	1·90	57	1170	360	540	900	—
d	2·17	1·90	225	1170	1425	540	1965	—
2d	3·97	2·21	282	1170	1530	295	1825	—
2e	3·97	2·21	282	308	1530	80	1610	—
e	2·17	1·90	215	308	1360	150	1510	—
f	2·17	1·90	280	1321	1775	610	2385	—
g	2·17	1·90	50	1321	315	610	925	—
3g	3·97	2·21	—	1321	—	335	335	—

Shear Stress Due to Bending

By inspection of the bending-moment diagram, it will be observed that at no point is the slope of the curve as great as the maximum slope in the overhang on the top rear spar,

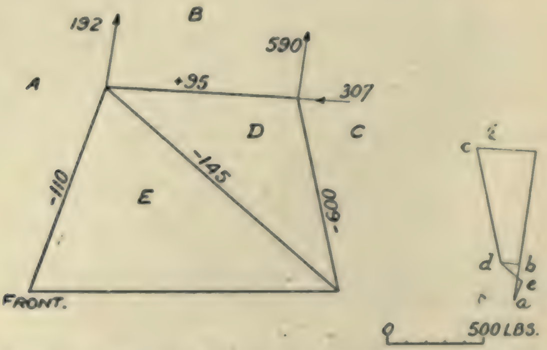


FIG. 20.—DRAG FORCES IN BODY STRUTS, C.P. AT ·5 OF CHORD. 5 TIMES LOAD

which means that the shear force is everywhere less in the bottom rear spar than in the top rear spar, and since the sections are identical, it follows that the shear stress at any point in the bottom spar is less than 385 lbs. per square inch, and is therefore quite allowable without further calculation.

Drag Forces on Top Plane

The resultant drag force on the bottom plane is, of course, taken at the fuselage direct, but the resultant drag force combined with the local lifts at the body strut joints are taken on the body struts and side bracing, as will be seen from Fig. 29, the resultant drag force being obtained from Fig. 24.

CHAPTER V

DOWN LOAD—CASE 3

An aeroplane in flight may sometimes be subjected to downward loading on the wing structure, and the circumstances under which such a condition of loading may exist are—

1. Possible downward gusts on the wings, or when looping.
2. Reversal of loading in a steep downward flight.

It is estimated that under the worst conditions of down loading, not more than about three times load is ever possible, hence the factor adopted is three times normal load with the C.P. assumed at about ·3 of the chord. It is very probable that in a vertical dive the C.P. is behind the rear spar, and the angle of incidence is small, which means that a down load on the front truss is experienced, and an up load on the rear truss. The amount of this loading is dependent upon the tail load, and the distance between the wing spars. For small machines, the load factor is about 3, decreasing for larger machines to about 1·5 times load.

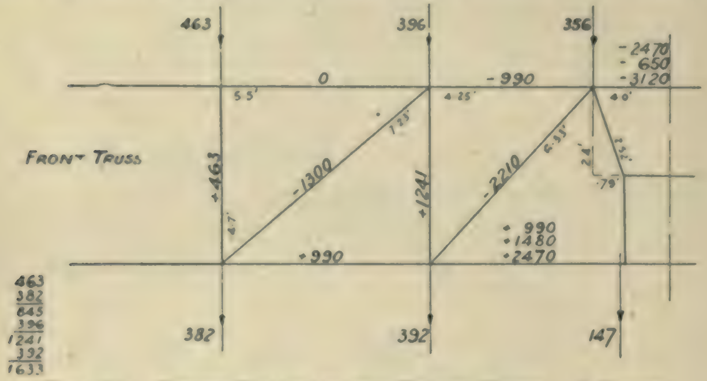
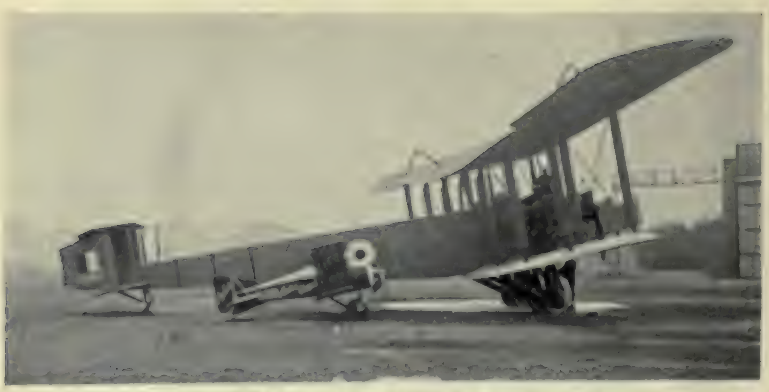


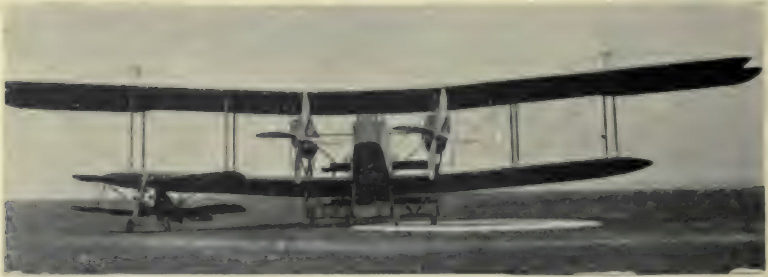
Fig. 30.—Force Diagram for Front Truss. 3 Times Down Load

It is usual to neglect the rear truss altogether. The rear anti-lift, or landing wires, must, however, be made of sufficient strength to withstand the worse case of landing, but these can be easily estimated.

The work in this case may be shortened considerably by a



HANDLEY-PAGE AND AUSTIN "WHIPPET"



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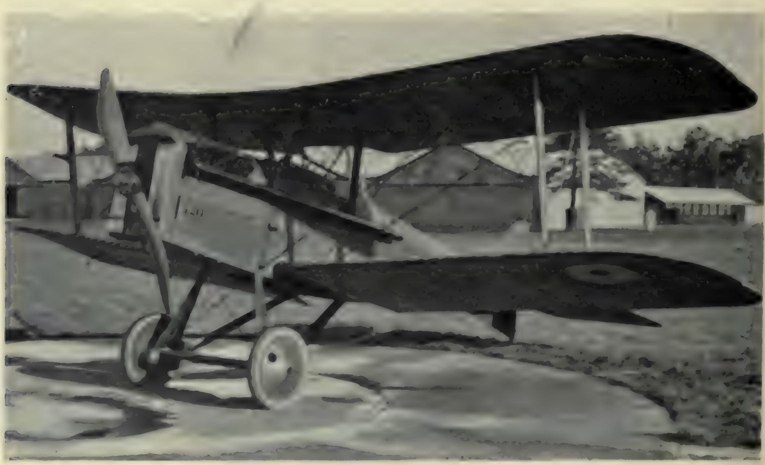

S.E.5 SINGLE-SEATER AEROPLANE

comparison with case (1) (*i.e.*, C.P. forward case), for it is possible to approximate very closely to the loading by so doing.

By reference to Fig. 11, and by multiplying the loads there by the ratio of the load factors in the two cases, namely, three-fifths, it is possible to determine the reactions in the down load case without troubling to draw a new load diagram.

It must be remembered, however, that the bottom spar is now in compression, and the top spar in tension, loads due to drag being omitted because their effect would be to reduce the compression in the bottom front spar. Fig. 30 shows the diagram of loads in the front truss bracing and spars, the reactions having been obtained by comparison with case (1).

At the head of Table 8 are given the particulars usually obtained from the load curve, which in this case have been obtained by comparison, as suggested above, and these are used in the determination of the final reactions for the top front spar. These final reactions are found to be very little different from the approximate reactions.

TABLE 8.

Point of Support 1	2	3		
Length of bay	5·5 ft.	4·25 ft.	2·0 ft. (Half Span)	
Load lbs. . 208	460	378	169	
Load per ft. . .	83·5	89·0	84·6	
$\frac{wl^2}{8}$ lbs. ft. . . .	316	201	169	
	$M_1 = -320$	$M_2 = -150$	$M_3 = -117$	
Span (ft.) . . .	5·5	4·25	4·0	
Span (ins.) . .	66	51	48	
I in.^{4}	4·77	4·77	4·77	
P_e	17400	29200	32900	$E = 1·6 \times 10^6$
End load P (ten.) .	0	990	3120	
$P_e + P$ . . .	17400	30190	36020	
w (lbs. per ft.) .	83·5	89·0	84·6	
$\frac{wl^2}{4}$	632	402	338	
$\frac{P}{P_e}$	0	·0339	·0948	
$1 - 0·12\frac{P}{P_e}$ . . .	1·000	·99593	·98862	
$1 + 0·32\frac{P}{P_e}$ . . .	1·000	1·0108	1·0303	
$1 + \frac{P}{80P_e}$ . . .	1·000	1·000424	1·001185	
$\frac{1}{(P_e + P)l}$ . . .	$1·045 \times 10^{-5}$	$0·78 \times 10^{-4}$	$0·695 \times 10^{-5}$	

This is now followed by the determination of the bending-moments at the various supports along the top front spar. The equations for M_{2_L} and M_{3_L} are from formula (7) in Appendix—

$$-320 \times 1{\cdot}0 \times 1{\cdot}045 \times 10^{-5} + 2M_{2_L} \times 1{\cdot}0 \times 1{\cdot}045 \times 10^{-5} + 632 \times 1{\cdot}0 \times 1{\cdot}045 \times 10^{-5} + M_{3_L} \times {\cdot}9959 \times {\cdot}78 \times 10^{-5} + 2M_{2_L} \times 1{\cdot}0108 \times {\cdot}78 \times 10^{-5} + 402 \times 1{\cdot}0004 \times {\cdot}78 \times 10^{-5} = O.$$

Dividing through by $\cdot 78 \times 10^{-5}$ this becomes

$$-429 + 2{\cdot}68M_{2_L} + 847 + {\cdot}9959M_{3_L} + 2{\cdot}0216M_{2_L} + 402 = O.$$

$$4{\cdot}7M_{2_L} + {\cdot}996M_{3_L} = -820 \quad . \quad . \quad . \quad (A)$$

$$M_{2_L} \times {\cdot}9959 \times {\cdot}78 \times 10^{-5} + 2M_{3_L} \times 1{\cdot}0108 \times {\cdot}78 \times 10^{-5} + 402 \times 1{\cdot}0004 \times {\cdot}78 \times 10^{-5} + M_{3_L} \times {\cdot}9886 \times {\cdot}695 \times 10^{-5} + 2M_{3_L} \times 1{\cdot}0303 \times {\cdot}695 \times 10^{-5} + 338 \times 1{\cdot}001185 \times {\cdot}695 \times 10^{-5} = O.$$

Dividing through by $\cdot 695 \times 10^{-5}$ this reduces to

$$1{\cdot}118\,M_{2_L} + 2{\cdot}27M_{3_L} + 451 + {\cdot}9886\,M_{3_L} \times 2{\cdot}0606\,M_{3_L} + 338 = O.$$

$$1{\cdot}118\,M_{2_L} + 5{\cdot}3192\,M_{3_L} = -789 \quad . \quad . \quad . \quad (B)$$

$$\begin{array}{lrcl}
(A) \div 4{\cdot}2 & 1{\cdot}118\,M_{2_L} + {\cdot}237\,M_{3_L} & = & -195 \\
 & 1{\cdot}118\,M_{2_L} + 5{\cdot}319\,M_{3_L} & = & -789{\cdot} \\
\hline
 & 5{\cdot}082\,M_{3_L} & = & -594{\cdot} \\
 & M_{3_L} & = & -117 \text{ lbs. ft.}
\end{array}$$

$$\text{From (A) } M_{2_L} = \frac{-820 + 116{\cdot}5}{4{\cdot}7}$$

$$= -150 \text{ lbs. ft.}$$

The corrected reactions are—

$$R_1 = 208 + 230 + \frac{320 - 150}{5{\cdot}5} = 470 \text{ lbs.}$$

$$R_2 = 230 + 189 - 32 + \frac{150 - 117}{4{\cdot}25} = 395 \text{ lbs.}$$

$$R_3 = 189 + 169 - 8 = 350 \text{ lbs.}$$

$$1215 \text{ lbs.}$$

At this point there would normally be inserted the calculations necessary for the estimation of the bending-moment at various points along the spar, but by inspection of the final bending-moments at the supports, and by comparison with those found in case (1) for the top front spar; and, remembering that the end load in this case is tensional as against compression in case (1), it will be recognized that it is unnecessary to draw the bending-moment diagram for this spar, because the stresses are certain to be less than in the former case.

Bottom Front Spar.

Similarly at the head of Table 9 will be found the particulars obtained by comparison with case (1), which are usually obtained from the load curve. Then follows the tabulation of quantities necessary for the finding of the final bending-moments and reactions at the supports, from which it will be observed that they differ very slightly from the approximate figures obtained by comparison.

TABLE 9.

Point of Support	1	2	3
Length of bay		5·5 ft.	4·95 ft.
Load lbs.	174	384	364
Load per ft.	—	69·8	73·5
$\frac{wl^2}{8}$ lbs. ft.	—	265	226

	$M_1 = -268$	$M_2 = -182$	$M_3 = 0$
Span (ft.)	5·5	4·95	
Span (ins.)	66	59·4	
I in.^{4}	4·77	4·77	
P_e	17400	21500	$E = 1{\cdot}6 \times 10^6$
End load P (comp.)	990	2470	
$P_e - P$	16410	19030	
w lbs. per ft.	69·8	73·5	
$\frac{wl^2}{4}$	530	452	
$\frac{P}{P_e}$	·0569	·1148	
$1 + 0{\cdot}2\frac{P}{P_e}$	1·0114	1·02296	
$1 - 0{\cdot}38\frac{P}{P_e}$	·9784	·9564	
$1 - \frac{P}{70P_e}$	·999187	·99836	

TABLE 9 (*continued*)

$\frac{1}{(P_e - P)l}$	$1{\cdot}108 \times 10^{-5}$	$1{\cdot}061 \times 10^{-5}$	
$M_R \quad M_L$	$-268 \quad -182$	$-182 \quad 0$	
$\frac{1}{2}(M_R + M_L)$	-225	-91	
$1 + {\cdot}26\frac{P}{P_e}$	1·0148	1·0299	
$1{\cdot}02\frac{wl^2}{8}$	270	231	
$M_L - M_R$	86	182	
$\frac{(M_L - M_R)^2}{2wl^2}$	1·745	9·15	

The equation for M_{2_L} is, from (1) in Appendix—

$$-268 \times 1{\cdot}0114 \times 1{\cdot}108 \times 10^{-5} + 2\,M_{2_L} \times {\cdot}9784 \times 1{\cdot}108 \times 10^{-5} + 530 \times {\cdot}9992 \times 1{\cdot}108 \times 10^{-5} + 2\,M_{2_L} \times {\cdot}9564 \times 1{\cdot}061 \times 10^{-5} = O.$$

$$\therefore\ 2{\cdot}045\,M_{2_L} + 1{\cdot}9128\,M_{2_L} = +283 - 554 - 451$$

$$3{\cdot}9578 M_{2_L} = -722$$

$$\underline{M_{2_L} = -182 \text{ lbs. ft.}}$$

The corrected reactions are—

$$R_1 = 174 + 192 + \frac{268 - 182}{5{\cdot}5} = 382 \text{ lbs.}$$

$$R_2 = 192 + 182 - 16 + \frac{182}{4{\cdot}95} = 395 \text{ lbs.}$$

$$R_3 = 182 - 37 = 145 \text{ lbs.}$$

$$\underline{922 \text{ lbs.}}$$

We must in this case perform the calculations required in the estimation of the bending-moment at several points along the spar, and completely draw a bending-moment diagram; for it will be remembered that since no bending-moment diagram was constructed for this spar in case (1), it is not possible to make a comparison as was done for the top front spar down load case.

Bending-Moments in Bays

Outer Bay—

$$M_{mid} = \frac{17400}{16410}(-225 \times 1{\cdot}0148 + 270)$$

$$= \frac{17400}{16410} \times 42$$

$$= \underline{44{\cdot}5 \text{ lbs. ft.}}$$

$$M_{max} = 44{\cdot}5 + 1{\cdot}745 \qquad = 45 \text{ lbs. ft.}$$

$$a = 2{\cdot}75 + \frac{86}{69{\cdot}8 \times 5{\cdot}5} = 2{\cdot}75 + {\cdot}224 = 2{\cdot}974 \text{ ft.}$$

$$c_1 = 2{\cdot}75 + {\cdot}9 \times {\cdot}224 - 1{\cdot}2\sqrt{\frac{45{\cdot}2}{69{\cdot}8}}$$

$$= 2{\cdot}75 - {\cdot}202 - {\cdot}965 = 1{\cdot}987 \text{ ft.}$$

$$c_2 = 2{\cdot}75 + {\cdot}202 + {\cdot}965 = 3{\cdot}917 \text{ ft.}$$

Inner Bay—

$$M_{mid} = \frac{21500}{19030}(-91 \times 1{\cdot}0299 + 231)$$

$$= \frac{21500}{19030} \times 137{\cdot}3$$

$$= \underline{155 \text{ lbs. ft.}}$$

$$M_{max} = 155 + 9{\cdot}15 \qquad = 164 \text{ lbs. ft.}$$

$$a = 2{\cdot}475 + \frac{182}{73{\cdot}5 \times 4{\cdot}95} = 2{\cdot}475 + {\cdot}5 = 2{\cdot}975 \text{ ft.}$$

$$c_1 = 2{\cdot}475 + {\cdot}9 \times {\cdot}5 - 1{\cdot}2\sqrt{\frac{164}{73{\cdot}5}}$$

$$= 2{\cdot}475 + {\cdot}45 - 1{\cdot}79 = 1{\cdot}135 \text{ ft.}$$

The values of the bending-moment at various intermediate points are now determined so that the bending-moment diagram which is shown in Fig. 31 may be drawn.

It will be observed that no offset B.M.s have been included in this case, since the anti-lift wires are assumed not to be offset at the points of support.

The points *a*, *b*, *d*, *e*, etc., are the points where spindling begins at 3 in. on either side each support, and the stresses are shown in Table 10, as has been done in the two previous cases, from which it will be seen that the spar is easily up to strength for this particular case.

The maximum slope of the bending moment diagram is less than that in C.P. forward, hence the shear stress is obviously satisfactorily small, and needs no further investigation.

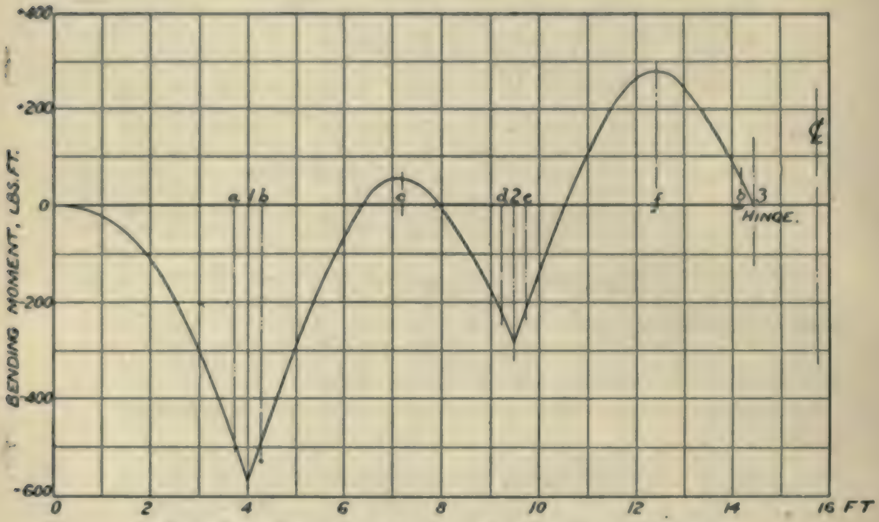

FIG. 31.—BENDING-MOMENT DIAGRAM FOR BOTTOM FRONT SPAR. 3 TIMES DOWN LOAD

TABLE 10

BOTTOM FRONT SPAR 3 TIMES DOWN LOAD

Section.	*A* in.^{2}	*Z* in.^{3}	B.M. lbs. ft.	End Load. lbs.	B.M. Stress. lbs./in.^{2}	E.L. Stress. lbs./in.^{2}	Total Stress. lbs./in.^{2}	F/S on 5500. lbs./in.^{2}
a	2·3	2·32	230	—	1190	—	1190	—
1*a*	4·8	3·17	268	—	1015	—	1015	—
1*b*	4·8	3·17	268	990	1015	205	1220	—
b	2·3	2·32	230	990	1190	430	1620	3·38
c	2·3	2·32	45	990	230	430	660	—
d	2·3	2·32	150	990	775	430	1205	—
2*d*	4·8	3·17	182	990	690	205	895	—
2*e*	4·8	3·17	182	2470	690	515	1205	—
e	2·3	2·32	140	2470	725	1070	1795	3·06
f	2·3	2·32	164	2470	850	1070	1920	2·86

Loads in Body Struts

The loads in the main anti-lift wire, and the down load reaction at that joint, are taken by the body struts and the side bracing between them.

Fig. 32 indicates the forces in the members concerned due to these loadings.

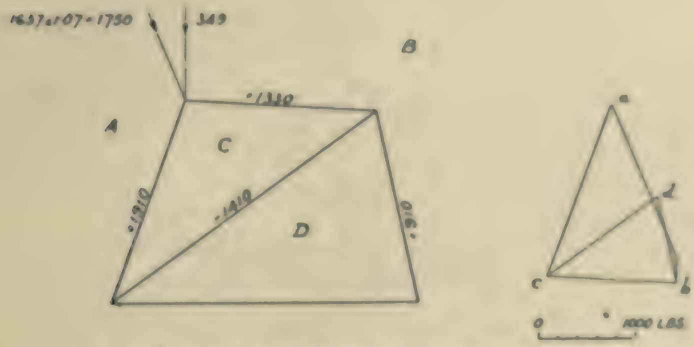

FIG. 32.—DRAG FORCES IN BODY STRUTS. 3 TIMES DOWN LOAD

True Loads

The true loads in the outer and inner anti-lift wires and also in the interplane struts are obtained approximately by multiplying the apparent forces in those members as shown in Fig. 30, by a factor of 1·1, which allows for the inclination to the vertical. True loads are given in Table 11.

CHAPTER VI

LIFT WIRES

There are two methods of duplicating lift wires, viz.—directly and indirectly.

Direct Duplication

This is carried out by using two separate wires in place of a single wire, each wire being of sufficient strength to carry the required load. In the event of one of these wires breaking, the remaining one enables the machine to fly to a suitable landing ground.

Hence, as the true loads in the front outer and main lift wires at five times load, when the C.P. is at ·3 of the chord, are 2490 lbs. and 3960 lbs. respectively, the sizes of wires would be—

Front outer lift = 2 – $\frac{1}{4}$ BSF wires (3450 lbs.)
Front main lift = 2 – 9-32 in. BSF wires (4650 lbs.)

The corresponding loads in the rear outer and main lift wires at five times load when the C.P. is at ·5 of the chord are 2720 lbs. and 4290 lbs. respectively, so that the same size wires could be used as for front truss. The incidence wires would be used mainly for erection purposes, and would not have to take such heavy loads as in indirect duplication.

Indirect Duplication

This is probably the better method and is accomplished by using a double lift wire as before, but possibly of smaller size, while the incidence wire is designed such that it can transfer loads from the front truss to the rear truss, or *vice versa*, in the case of a lift wire failing completely.

It is usual to make each of the twin lift wires strong enough to take two-thirds of the load normally taken by the single non-duplicate wire.

For example, the true maximum loads taken by the front outer and main lift wires are 2490 lbs. and 3960 lbs. respectively at five times load when the C.P. is at ·3 of the chord.

For this method of duplication two separate wires must be inserted in the outer bay, each capable of taking two-thirds of 2490 lbs., viz., 1660 lbs., and the main lift wire replaced by two wires each capable of taking two-thirds of 3960 lbs., viz., 2640 lbs. Hence the wires to be used to correspond with these loads are—

Front outer lift = 2 – 2 BA wires (1900 lbs.)
Front main lift = 2 – $\frac{1}{4}$ in. BSF wires (3450 lbs.)

Similarly the maximum true loads taken by the rear outer and main lift wires at five times load, when the C.P. is at ·5 of the chord, are 2720 lbs. and 4290 lbs. respectively. Taking two-thirds of these loads (*i.e.*, 1810 lbs. and 2860 lbs.), the same sizes of wires can be used as for the front truss, namely—

Rear outer lift = 2 – 2 BA wires
Rear main lift = 2 – $\frac{1}{4}$ BSF wires

When the front outer lift wire fails completely, the true load in the forward incidence wire is 1430 lbs. (Fig. 33), and it must, therefore, be 2 BA (1900 lbs.).

In the case of the rear outer lift wire failing completely, the true load in the backward incidence wire is 1123 lbs (Fig. 34), so that the wire required is 2 BA (1900 lbs.).

When loads are transferred from one truss to another in this way, increased loads may be imposed on certain members, as explained in the wires broken-case.

CASE 4

Wires Broken Case. 3·5 Times Load

It is customary nowadays to design the wing structure of an aeroplane so that in case a lift wire breaks the structure still has a factor of safety, which will enable it, at least, to maintain steady flight for the time being, in order that a safe landing may be assured at the earliest opportunity. The factor usually demanded is about two-thirds of that allowed in normal flight (*i.e.*, two-thirds of 5, or say $3\frac{1}{2}$ times load), and this is the factor to be adopted in the following calculations.

In the ordinary cases of flying already considered, when the lift wires were unbroken nothing was discussed as to the possible function of the incidence bracing, for it was difficult to estimate to what extent these incidence wires could take load. In this case, however, where either the front or the rear lift wire is broken, it will be clear that if the structure is to withstand the flying loads, then the incidence bracing must and will become operative. Any flying wire may possibly be broken, and actually each case ought to be considered separately.

Since, however, the application of the principle is similar in each case, it is only proposed to indicate the procedure in two of the possible cases.

(a) Front Outer Lift Wire Broken, C.P. at ·3 of the Chord

When the front outer lift wire is broken, the result is that none of the load at the top and bottom front outer joints can be carried by the front main lift wire, but must be transferred through the forward incidence wire to the bottom rear outer strut joint, thence along the rear truss in the ordinary way. This reduces to the consideration of the rear truss, with the centre of pressure forward, together with an added load at the bottom outer strut joint, which has been transferred from the front truss, the factor of loading being 3·5 times load.

A line diagram is drawn showing the rear truss at the top of the sheet, and the front truss at the bottom, with plans of the top and bottom wing bracing, as shown in Fig. 33, where the loads indicated in the front and rear trusses are $\frac{3{\cdot}5}{5{\cdot}0} = {\cdot}7$ times those from Fig. 11 for C.P. forward.

Referring to the front truss, since the front outer lift wire is broken, then the two loads, of magnitude 447 lbs. and 541 lbs. respectively, are transferred from the front truss to the bottom outer point of the rear truss as shown, and added to the load 216 lbs. already existent there, the other loads all being ·7 times those obtained previously in case (1) for C.P. forward.

Drag Forces

Some amount of care must be exercised in the estimation of the drag forces at the bottom rear outer and the top front outer joints. Consider the top outer strut joint of the front truss where the forces at the joint are: resultant lift 541 lbs., load up the strut 447 lbs. load in forward incidence wire, and drag force on wing, these two latter being unknown.

The drag force on the wing (there being no immediate need to find the other unknown force) is found by using the drag coefficients from Fig. 14; its magnitude is—

$$0{\cdot}912 \times 541 + ({\cdot}912 - {\cdot}335)\,447 = 493 + 258 = 751 \text{ lbs.}$$

from front to rear.

In a similar manner, the drag force at the bottom rear outer strut joint is determined, the forces at that joint being: resultant lift, load in incidence wire, load in strut, drag force on wing.

The drag force is found by again making use of the drag coefficients from Fig. 14—

$= {\cdot}335 \times 216$ from front to rear $= 72$ lbs.

and

$(541 + 447)\,({\cdot}912 - {\cdot}335)$ from rear to front $= 570$ lbs.

giving a drag force at that point of $\underline{498}$ lbs.

acting from rear to front.

The drag forces at the other points present no real difficulty, and are not fully worked out here, but a reference to Fig. 33 will show how they have been obtained.

It will be noticed that the resultant drag on the top wing amounts to only 50 lbs. from rear to front, to be taken on the side bracing and rear body struts, but this is so small compared with the resultant drags obtained in the cases already considered, that it is quite unnecessary to find the loads in those members for this case.

End Loads in Spars

Having obtained all the drag forces at the several joints, top and bottom wings, the loads in the drag bracing and the end loads in the spars are calculated, or obtained graphically. The mean end load in each bay due to drag is added to that in the same bay due to lift, as shown in Fig. 33.

To estimate the load in the incidence wire, which transfers

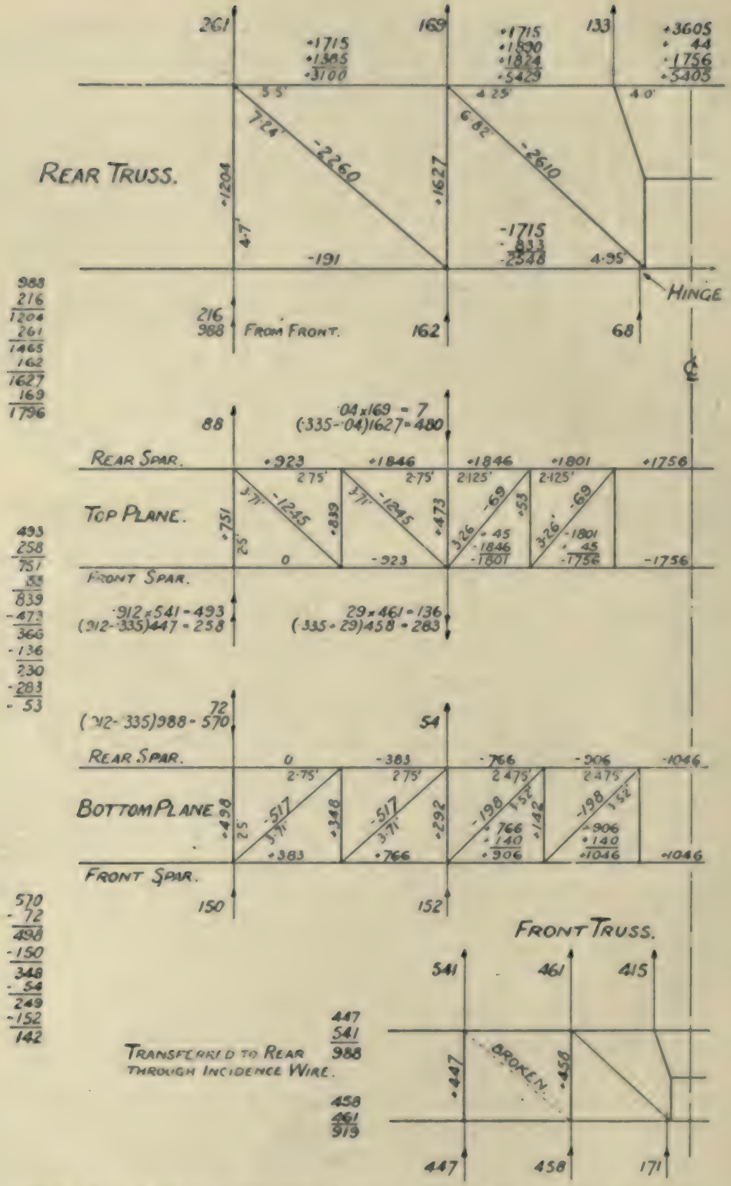

FIG. 33.—FORCE DIAGRAM, FRONT OUTER LIFT WIRE BROKEN, C.P. AT ·3 OF THE CHORD. 3·5 TIMES LOAD

the load from top front to bottom rear, the coefficient from Fig. 14 must be used; *i.e.*, load in incidence wire

= 988 × 1·445 = 1430 lbs.

It will furthermore be observed that the end loads in the spars are about the same, or rather less than those for the rear truss (C.P. at ·5, case 2), and since the bending stresses are approximately ·7 of those in that same case, then the rear spars are obviously up to strength, and need not concern us further for this case of front outer lift wire broken. A few members, however, have sustained heavier loads, namely, the load in the rear outer strut and a few compression ribs; consequently, these are probably the loads which determine the size of these members.

(b). Rear Outer Lift Wire Broken, C.P. at ·5 of the Chord

In this case, since the rear outer lift is incapable of transferring the load from the outer strut joints, this load must be transferred from rear to front by means of the backward incidence wire. The factor to be adopted again is 3·5 times load, hence a diagram of loads must be made, the loads being ·7 of those for case (2) C.P. rear, and thus this case reduces to a consideration of the front truss, C.P. rear, with additional load transferred from the rear. Fig. 34 shows this diagram with the loads inserted thereon.

Drag Forces

Just as in the case of front outer lift wire broken, some amount of care must be exercised in the estimation of the drag forces, particularly at the bottom front outer and the top rear outer joints, where the load in the incidence wire affects the drags, otherwise the work is very similar to that in case (*a*) just considered.

Here, again, the end loads are not greatly different from those obtained for the front truss, C.P. at ·3 times chord, and since the bending stresses are approximately ·7 of these obtained in that case, then the spars are obviously up to strength. The front outer strut and lift wire sustain somewhat greater

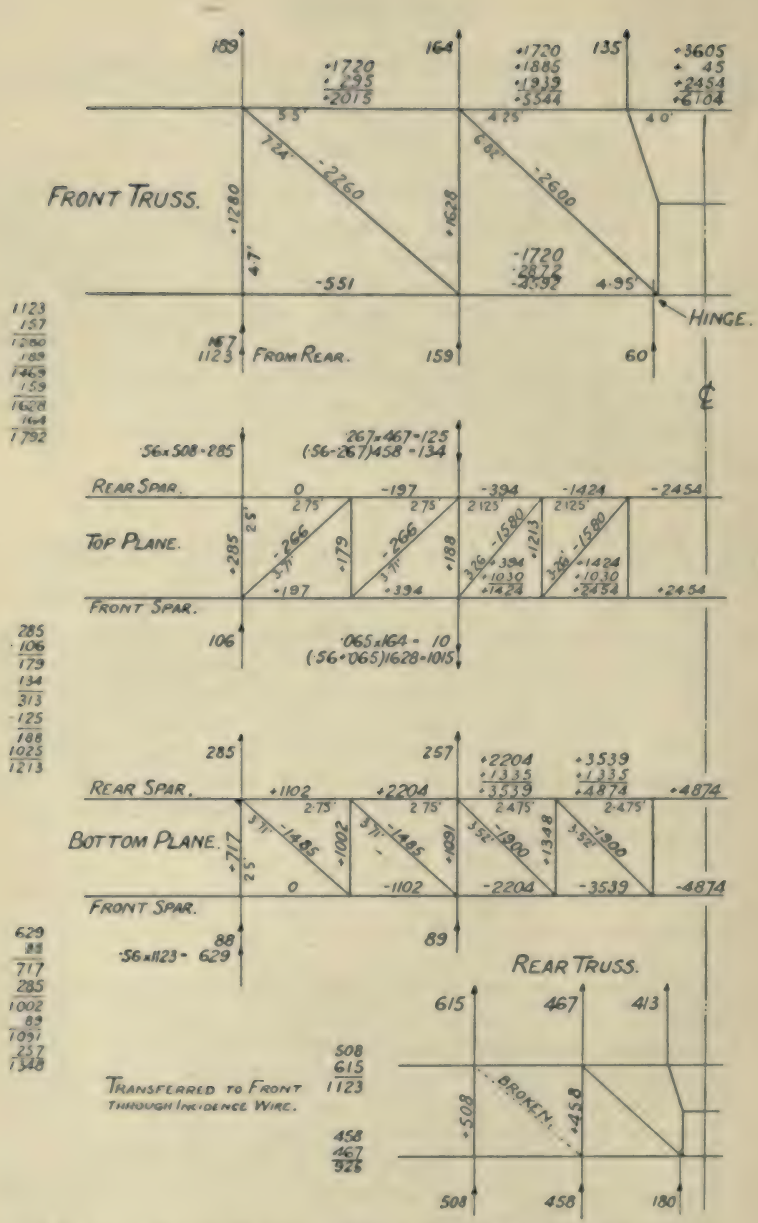

FIG. 34.—FORCE DIAGRAM. REAR OUTER LIFT WIRE BROKEN, C.P. AT ·5 OF THE CHORD. 3·5 TIMES LOAD

loads, and these are the loads which affect the design of these members.

It will be noticed in Fig. 34 that the resultant drag load on the top wing is 1213 lbs., and this is to be taken on the front body strut and the forward bracing wire. The force diagram is shown in Fig. 35.

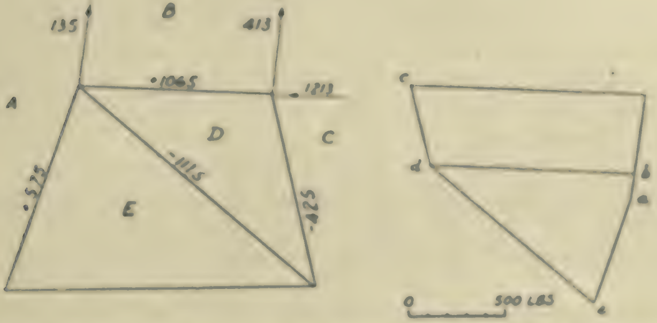

FIG. 35.—DRAG FORCES IN BODY STRUTS, REAR OUTER LIFT WIRE BROKEN; C.P. AT ·5 OF THE CHORD 3·5 TIMES LOAD

(c) General Condition

By taking the drag and lift loads from either Fig. 33 or Fig. 34 and applying them directly to the body struts and side bracing, we have assumed that the same wire is broken on each side of the machine. As this possibility is somewhat remote, the correct procedure would be to make a diagram of the centre section including the body struts and wires and apply the loads which would most likely occur. Thus, suppose as in case (*a*) the front outer lift wire is broken on the port side of the machine while the wires on the starboard side are all intact. The end loads in the top spars at the body strut joints and the load in the drag wire are obtained immediately from Fig. 33 for the port side; for the starboard side the corresponding loads will obviously be ·7 of those shown in Fig. 11. The resultant end loads in the front and rear spars will not be quite equal, but should balance out with the end loads in the bottom spars. It is thus seen that all the cabane members will carry loads,* the most important of which are the body struts with their cross bracing and the centre section internal cross bracing.

The approximate loads, may, however, be obtained quickly. The resultant spar end loads (*i.e.*, including the component drag wire load along the direction of the spar) being about 5,000 lbs. it would suffice to make both the internal bracing and cross bracing single ¼ in. B.S.F. wires, as they would have to carry about 3,000 lbs. each. It is also fairly clear that the maximum body strut load will be of the order of 2,000 lbs. for which load these members are designed. In the example we have just considered, there is body strut bracing in both front and rear bays. Should, however, only one bay be braced, it may be necessary to duplicate the internal bracing. In any case the fuselage bulkhead bracing at the body strut attachments should not be overlooked, as the loads on the top and bottom wings must, in general, balance out through these wires.

Similarly if case (*b*) were considered, the loads being obtained from Figs. 14 and 34 it would be found that approximately the same loads would be obtained in the several members.

Having now completed the several cases, namely—

C.P. forward, 5 times load
C.P. back 5 ,, ,,
C.P. forward, 3 times down load

and the two wires broken cases, it is well to draw up a table and a diagram in conjunction with it, indicating the true loads in all the struts and wires of the wing structure.

The true loads may be obtained by correcting the inclination of the struts and wires to the vertical by utilizing the factors given on the drag coefficient diagrams for the particular cases concerned.

This has been done in Table 11 and the accompanying diagram where the loads indicated are the true loads in every case.

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* The actual load in each member can be obtained by the application of the Principle of Least Work. This subject is well treated in *Aeroplane Structures* by Pippard and Pritchard. (Longmans, Green & Co.)

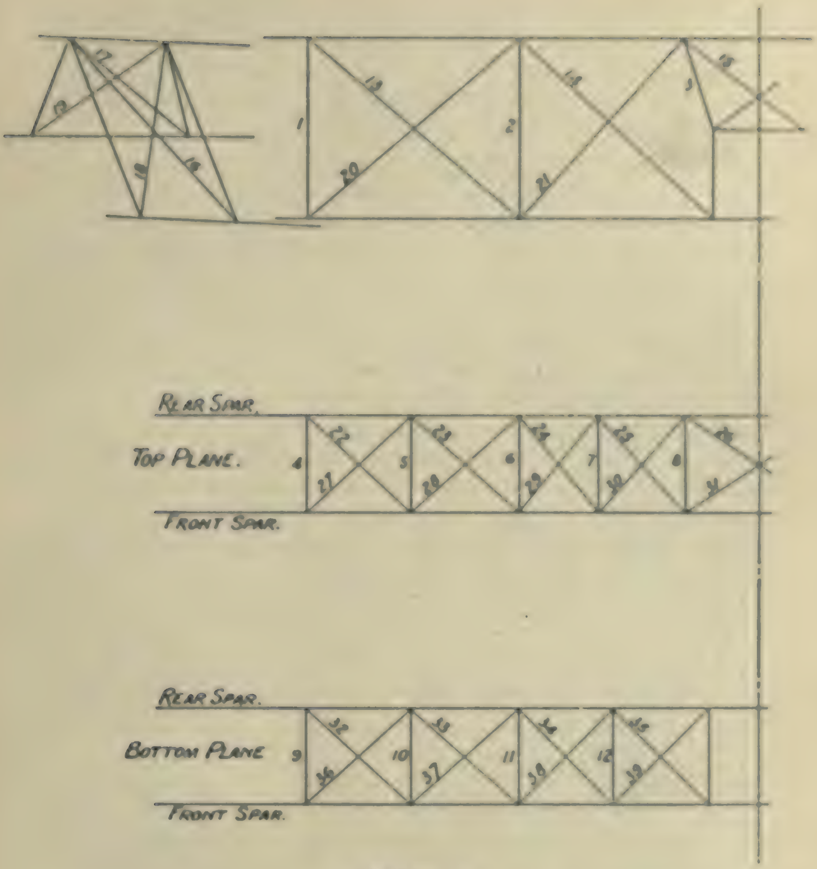


TABLE 11

	Member.	Size of wire or failing load.	5 times load, C.P. at .3 of the chord.	5 times load, C.P. at .5 of the chord.	3 times down load.	Wire broken, C.P. at .3 of the chord.	Wire broken, C.P. at .5 of the chord.	Maximum load.
Interplane Struts.	1*F*	—	700	—	510	—	1410	1410
	1*R*	—	—	800	—	1325	—	1325
	2*F*	—	2265	—	1360	—	1790	2265
	2*R*	—	—	2480	—	1790	—	2480
	3*F*	—	450	− 110	1910	− 460	575	1910 −460
	3*R*	—	− 195	− 600	910	—	− 425	910 −600

TABLE 11 (*continued*)

	Member.	Size of wire or failing load.	5 times load, C.P. at ·3 of the chord.	5 times load, C.P. at ·5 of the chord.	3 times down load.	Wire broken, C.P. at ·3 of the chord.	Wire broken, C.P. at ·5 of the chord.	Maximum load.
Drag Struts.	4	—	259	151	—	751	285	751
	5	—	384	642	—	839	179	839
	6	—	259	483	—	473	188	483
	7	—	1355	307	—	53	1213	1355
	8	—	1325	95	1330	—	1045	1330
	9	—	213	125	—	498	717	717
	10	—	213	531	—	348	1002	1002
	11	—	533	658	—	292	1091	1091
	12	—	611	1024	—	142	1348	1348
Wing Bracing.	13*F*	2–2 B.A.	–2390	—	—	—	–2490	–2490
	13*R*	2–2 B.A.	—	–2720	—	–2490	—	–2720
	14 *F*	2-¼in.B.S.F	–3950	—	—	—	–2620	–3950
	14 *R*	2-¼in.B.S.F	—	–4290	—	–2630	—	–4290
	15 *F*	1-¼in.B.S.F	—	—	—	–3000	–3000	–3000
	15 *R*	1-¼in,B.S.F	—	—	—	–3000	–3000	–3000
	16	2 B.A.	—	—	—	–1430	—	–1430
	17	2 B.A.	–1620	– 145	—	—	–1115	–1620
	18	2 B.A.	—	—	—	—	–1123	–1123
	19	2 B.A.	—	—	–1410	—	—	–1410
	20 *F*	2 B.A.	—	—	–1430	—	—	–1430
	20 *R*	2 B.A.	—	—	—	—	—	—
	21 *F*	¼in. B.S.F.	—	—	–2430	—	—	–2430
	21 *R*	2 B.A.	—	—	—	—	—	—
Drag Bracing in Wings.	22	2 B.A.	– 570	– 951	—	–1245	—	–1245
	23	2 B.A.	– 570	– 851	—	–1245	—	–1245
	24	2 B.A.	—	—	—	—	—	—
	25	2 B.A.	—	—	—	—	—	—
	26	1-¼in.B.S.F	—	—	—	—	–3000	–3000
	27	4 B.A.	—	—	—	—	– 266	– 266
	28	4 B.A.	—	—	—	—	– 266	– 266
	29	2 B.A.	–1765	– 400	—	– 69	–1580	–1765
	30	2 B.A.	–1765	– 400	—	– 69	–1580	–1765
	31	1-¼in.B.S.F	—	—	—	–3000	—	–3000
	32	2 B.A.	– 469	– 787	—	—	–1485	–1485
	33	2 B.A.	– 469	– 787	—	—	–1485	–1485
	34	2 B.A.	– 860	–1444	—	—	–1900	–1900
	35	2 B.A.	– 860	–1444	—	—	–1900	–1900
	36	4 B.A.	—	—	—	– 517	—	– 517
	37	4 B.A.	—	—	—	– 517	—	– 517
	38	4 B.A.	—	—	—	– 198	—	– 198
	39	4 B.A.	—	—	—	– 198	—	– 198

From the column of maximum loads in the above table the sizes of struts and wires are finally determined.

A table of sizes of streamline wires in general use, and the corresponding strengths, is given in Table 30.

CHAPTER VII

DESIGN OF STRUTS

The design of tapered struts has occupied the energies of mathematicians for many years, and it is only recently that the correct mathematical solution has been introduced, and made of practical use to aeroplane designers and others. This solution is due to the exhaustive work of Messrs. Webb and Barling, who read a paper before the Royal Aeronautical Society and published it in the official organ of that Society in October, 1918, where the mathematical solution and its applications to design may be found, and to which reference is now suggested.

Generally, external struts are of streamline section, either solid or hollow, wood or metal. However, in these calculations it is assumed that solid wooden struts are to be used, and the following are the necessary calculations for the outer, inner, and body struts respectively.

These struts are designed to take a load which in each case is slightly greater than the maximum load shown in Table 11, the section being solid streamline, having a fineness ratio of 3 to 1 (*i.e.*, having a length three times its breadth).

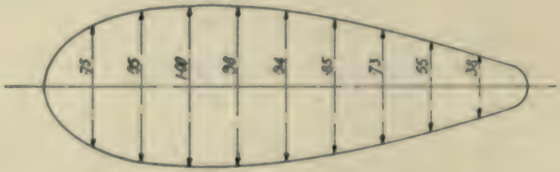
For a streamline section of fineness ratio 3 to 1, the least moment of inertia is given by $I = \cdot 13\, t^4$ where t is the maximum thickness, while the area of the section is given by $A = 2.18 t^2$ (see Fig. 36). The material used is spruce, having the following characteristics—

Density $\rho = \cdot 018$ lbs. per in^3.
Young's Modulus $E = 1.6 \times 10^6$ lbs. per in^2.
Allowable compressive stress $f = 5{,}500$ lbs. per in^2.

Pin jointed ends are assumed, and the lengths quoted are the distances between the pin centres.

For meaning of symbols reference should be made to Fig. 37.

The advantage of using tapered struts is that they effect quite a considerable saving in weight for a given strength as compared with parallel streamline struts. Some designers, for the convenience of manufacture, prefer to taper their struts in ways not quite in accordance with that recommended by this method.



$A = at^2 \qquad I = bt^4$

FINENESS RATIO	a	b
2·5 : 1	1·82	·108
3·0 : 1	2·18	·130
3·5 : 1	2·54	·152
4·0 : 1	2·90	·173

FIG. 36.—STANDARD STRUT SECTION

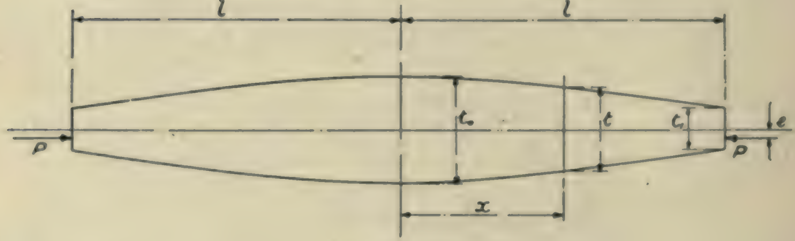


FIG. 37

Outer Interplane Struts

Both front and rear struts are to be designed for a load of 1450 lbs.

Length L between pin centres = 57 ins.

$\therefore t_o$ is approximately given by—

$$P = \cdot 8\, P_e = \frac{\cdot 8 \pi^2 E I_o}{L^2}$$

$$\therefore I_o = \frac{1450 \times 57^2}{\cdot 8 \times \pi^2 \times 1 \cdot 6 \times 10^6} = 0 \cdot 37$$

$$\therefore t_o^4 = \frac{I_o}{\cdot130} = \frac{\cdot37}{\cdot130} = 2{\cdot}85$$

$$\therefore t_o = 1{\cdot}30 \text{ in.}$$

$$\therefore A = 2{\cdot}18\, t_o^2 = 3{\cdot}68 \text{ in.}^2$$

$$\therefore \frac{P}{fA_o} = \frac{1450}{5500 \times 3{\cdot}68} = 0{\cdot}0716$$

From curve (Fig. 38) this gives $\frac{P}{P_o} = 0{\cdot}765$ instead of approximate value $= 0{\cdot}8$.

$$\therefore \text{Correct } I_o = 0{\cdot}387$$

$$t_o^4 = \frac{0{\cdot}387}{\cdot13} = 2{\cdot}98, \quad \therefore t_o = \underline{1{\cdot}32 \text{ in.}}$$

$$\therefore \qquad A_o = 3{\cdot}8 \text{ in.}^2 \text{ and } \frac{P}{fA_o} = \frac{1450}{5500 \times 3{\cdot}8} = 0{\cdot}0693$$

$$\therefore \qquad \frac{t_1}{t_o} = \sqrt{\cdot0693} = \cdot263$$

Then from curve (Fig. 39) for

$\frac{x}{l}$ =	0	·4	·5	·6	·7	8	·9	1·0

we have—

$\frac{t}{t_o}$ =	1	·94	·904	·866	·81	·732	·60	·263

and since $t_o = 1{\cdot}32$ in. and $l = 28{\cdot}5$ in.,

t in. =	1·32	1·24	1·19	1·15	1·07	0·97	0·79	0·35
x in. =	0	11·4	14·25	17·1	19·95	22·8	25·65	28·5

which gives shape of strut.

The Weight of Strut

$\frac{P}{fA_o} = \cdot0693$ $\therefore$ weight coefficient from curve (Fig. 38) $= 0{\cdot}90$, so that weight of strut $= \cdot758 \times \cdot018 \times 3{\cdot}8 \times 57 = 2{\cdot}95$ lbs.

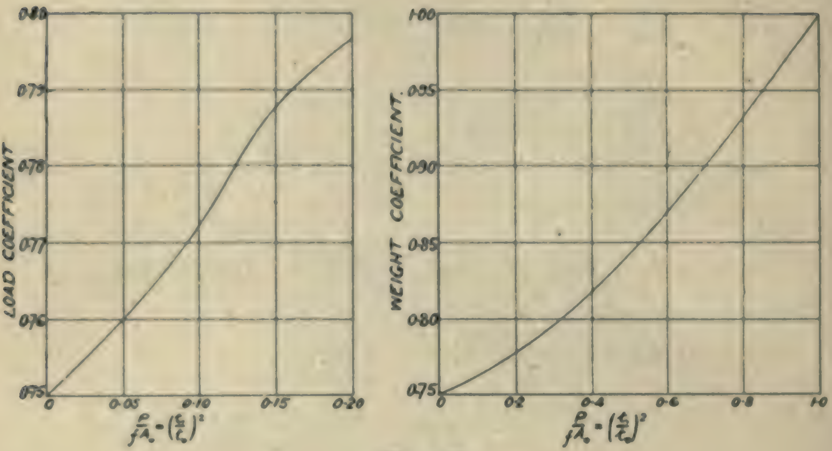


FIG. 38

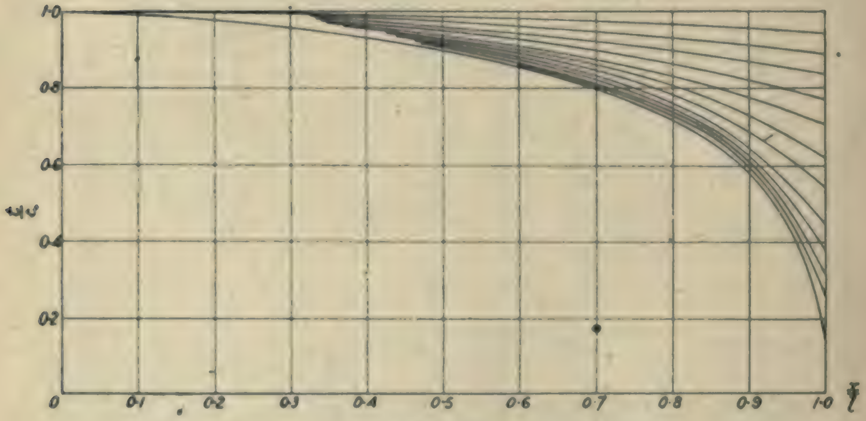


FIG. 39

Inner Interplane Struts

Both front and rear struts are to be designed for a load of 2500 lbs. Length L between pin centres = 57 ins.

$\therefore$ t_o is approximately given by—

$$P = \cdot 8 \, P_e = \frac{\cdot 8\pi^2 \, 1\cdot 6 \times 10^6 \, I_o}{L^2}$$

$$\therefore I_o = \frac{2500 \times 57^2}{{\cdot}8\pi^2 \times 1{\cdot}6 \times 10^6} = 0{\cdot}637$$

$$\therefore t_o^4 = \frac{{\cdot}637}{{\cdot}13} = 4{\cdot}91$$

$$\therefore t_o = 1{\cdot}49 \text{ in.}$$

$$A_o = 2{\cdot}18 \times 1{\cdot}49^2 = 4{\cdot}83 \text{ in.}^2$$

$$\frac{P}{fA_o} = \frac{2500}{5500 \times 4{\cdot}83} = 0{\cdot}0941$$

From curve (Fig. 38) $\frac{P}{P_e}$ is ·771

$$\therefore \text{Correct } I_o = 0{\cdot}662 \text{ in.}^4$$

$$\therefore t_o^4 = \frac{{\cdot}662}{{\cdot}13} = 5{\cdot}09 \qquad \underline{\therefore t_o = 1{\cdot}50 \text{ in.}}$$

$$A_o = 4{\cdot}92 \text{ in.}^2; \frac{P}{fA_o} = \frac{2500}{5500 \times 4{\cdot}92} = 0{\cdot}0925$$

$$\therefore \frac{t_1}{t_o} = \sqrt{{\cdot}0925} = {\cdot}304.$$

Then from curve (Fig. 39), for

$$\frac{x}{l} = 0 \quad {\cdot}4 \quad {\cdot}5 \quad {\cdot}6 \quad {\cdot}7 \quad {\cdot}8 \quad {\cdot}9 \quad 1{\cdot}0$$

we have

$$\frac{t}{t_o} = 1 \quad {\cdot}95 \quad {\cdot}92 \quad {\cdot}87 \quad {\cdot}82 \quad {\cdot}74 \quad {\cdot}61 \quad {\cdot}304$$

and since $t_o = 1{\cdot}50$ and $l = 28{\cdot}5$ in.,

t in. = 1·50 1·43 1·38 1·31 1·23 1·11 ·92 ·46

x in. = 0 11·4 14·25 17·1 19·25 22·8 25·65 28·5

which gives shape of strut.

Weight of Strut

Weight coefficient from curve (Fig. 38) is ·761.

Weight is ·761 × ·018 × 4·92 × 57 = 3·85 lbs.

Front Body Strut

Strut to be designed for a load of 2000 lbs. Length L between pin centres = 30 ins.

$\therefore t_o$ is approximately given by—

$$P = \cdot 8\, P_e = \frac{\cdot 8\pi^2 \times 1\cdot 6 \times 10^6\, I_o}{L^2}$$

$$\therefore I_o = \frac{2000 \times 30^2}{\cdot 8\pi^2 \times 1\cdot 6 \times 10^6} = \cdot 1425$$

$$\therefore t_o^4 = \frac{\cdot 1425}{\cdot 13} = 1\cdot 095 \text{ in.}$$

$$A_o = 2\cdot 18 \times 1\cdot 075^2 = 2\cdot 61 \text{ in.}^2$$

$$\frac{P}{fA_o} = \frac{2000}{5500 \times 2\cdot 61} = 0\cdot 139$$

From curve (Fig. 38) this gives $\dfrac{P}{P_e} = \cdot 785$ instead of $\cdot 8$.

$$\therefore \text{Correct } I_o = \cdot 145$$

$$\therefore t_o^4 = \frac{\cdot 145}{\cdot 13} = 1\cdot 115 \qquad \therefore t_o = \underline{1\cdot 03 \text{ in.}}$$

$$A_o = 2\cdot 18 \times 1\cdot 03^2 = 2\cdot 31 \text{ in.}^2 \text{ and}$$

$$\frac{P}{fA_o} = \frac{2000}{5500 \times 2\cdot 31} = \cdot 1575$$

$$\therefore \frac{t_1}{t_o} = \sqrt{\cdot 1575} = \cdot 397$$

Then from curve (Fig. 39), for

$$\frac{x}{l} = \quad 0 \quad \cdot 4 \quad \cdot 5 \quad \cdot 6 \quad \cdot 7 \quad \cdot 8 \quad \cdot 9 \quad 1\cdot 0$$

we have

$$\frac{t}{t_o} = \quad 1 \quad \cdot 95 \quad \cdot 92 \quad \cdot 88 \quad \cdot 83 \quad \cdot 75 \quad \cdot 64 \quad \cdot 397$$

and since $t_o = 1\cdot 03$ and $l = 15$ in.,

t in. =	1·03	·98	·95	·91	·85	·77	·66	·41
x in. =	0	6·0	7·5	9·0	10·5	12·0	13·5	15·0

Weight of Strut

Weight coefficient from curve (Fig. 39) is ·770

∴ Weight is ·77 × ·018 × 2·31 × 30 = 0·96 lbs.

The rear body strut could be made the same size.

FORMS OF HOLLOW STRUTS

If in the design of struts for large machines tapered streamline hollow wooden struts be used, the section can be determined by a method described in the same paper. Hollow wooden struts must of necessity be built up of several pieces. Fig. 40 shows two forms of section. There are also one or two patent methods of constructing built up struts.

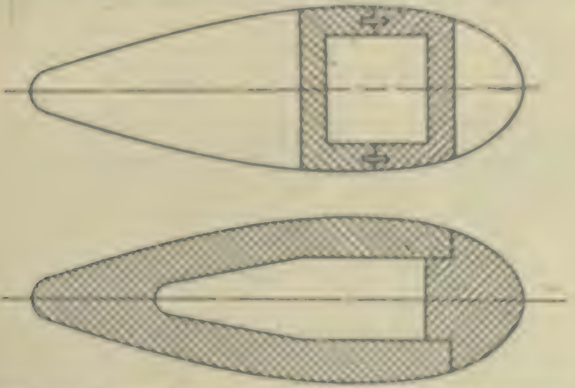


Fig. 40.—Two Forms of Hollow Strut Sections

Tapered metal tubular struts are rarely used, because of the difficulty of manufacture, but otherwise they possess advantages over wooden ones. Parallel streamline tubular struts, however, are often used, but they have the disadvantage of being heavy. Circular tubes with wooden fairing are much lighter.

In estimating the strength of streamline struts, it is useful to know that there is an approximately constant ratio between circular section struts and streamline struts where the material and gauge correspond.

For example, a streamline tube of fineness ratio 3 to 1 and whose minor diameter is *d*, has the same compressive strength as a circular tube of diameter 1·5 *d*, the gauge and length being the same.

LATERAL FAILURE OF SPARS

Spars tend to fail laterally (*i.e.*, about their major axes) as struts, due to compressive end load. In the case of a single bay of drag bracing between the interplane strut attachments the front and rear spars will bend independently, except for the ribs, tending to make both spars deflect the same amount. The spars are, however, stiffened against lateral failure if there are two bays of drag bracing between the two points of support, which are the strut attachments. When such is the case, the front and rear spars bend together, so that the crippling load depends on the sum of the lateral moments of inertia of the front and rear spar sections, the length being the distance between the two compression rib attachments.* (*See* Fig. 41.)

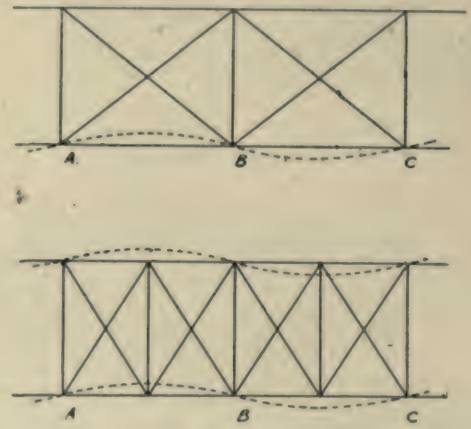


FIG. 41

The crippling load of such a bay would, therefore, be—

$$P_{e_F} + P_{e_R} = \frac{\pi^2 E(I_F + I_R)}{l^2}$$

where I_F and I_R are the least moments of inertia of the front and rear spars respectively, and l = length of bay in inches.

Undoubtedly, the effect of the fabric on secondary failure is considerable in small machines, but its extent is difficult to estimate.

The comparison between the crippling load, as above, and

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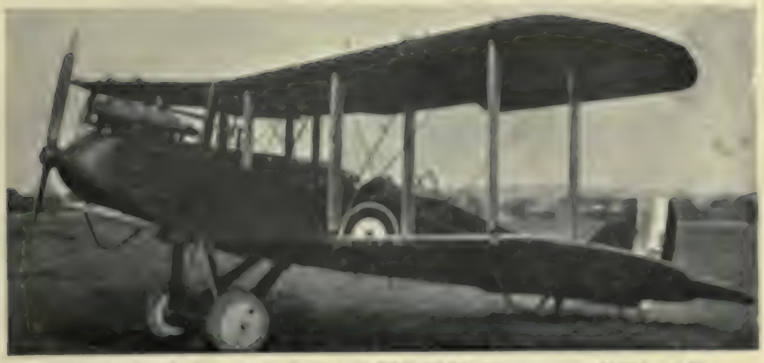
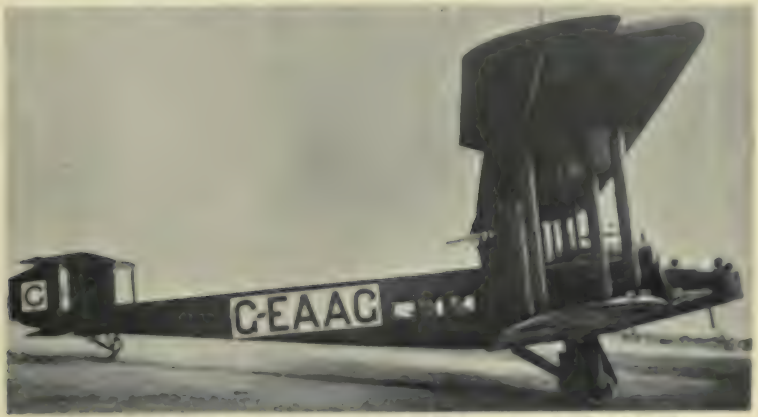
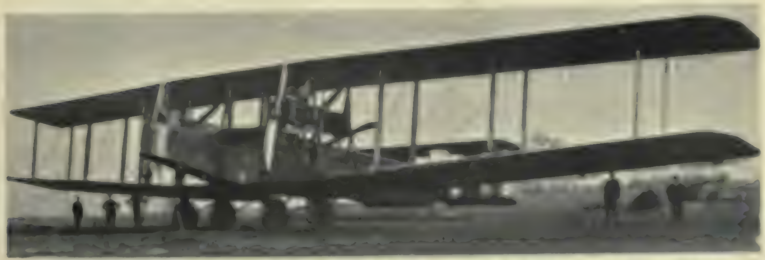
* This has been shown to be the case by Messrs. Booth and Bolas, and is treated more fully in *Aeroplane Structures*, by Pippard and Pritchard. (Longmans, Green & Co.)



AIRCO 9, TWO-SEATER, WITH 250 H.P. SIDDELEY "PUMA" ENGINE



HANDLEY-PAGE TWIN-ENGINED PASSENGER AEROPLANE STARTING UP FOR LONDON–PARIS–BRUSSELS SERVICE



HANDLEY-PAGE, TYPE V 1500, WITH 4–375 H.P. ROLLS-ROYCE "EAGLE" ENGINES. ORIGINALLY BUILT FOR LONG-DISTANCE BOMBING

the sum of the actual end loads in the front and rear spars for any one position of the centre of pressure, represents to some extent the factor of safety. This factor is at least 1·0 on five times load, as will be seen from the following calculations, although a factor of about 0·8 would be allowable.

Box spars, having larger lateral moments of inertia than ordinary I sections, are consequently better able to withstand the tendency to lateral failure.

Case 1. Flying with 5 Times Load, C.P. at ·3 of the Chord.

Top Outer Bay.

The end loads in the spars are 1440 lbs. and 1683 lbs. for front and rear respectively.

For lateral failure,

$$\text{Front spar } P_{c_F} = \frac{\pi^2 \times 1{\cdot}6 \times 10^6 \times {\cdot}213}{33^2} = 3120 \text{ lbs.}$$

$$\text{Rear spar } P_{c_R} = \frac{\pi^2 \times 1{\cdot}6 \times 10^6 \times {\cdot}24}{33^2} = 3520 \text{ lbs.}$$

$$\frac{P_{c_F} + P_{c_R}}{P_F + P_R} = \frac{3120 + 3520}{1440 + 1683} = 2{\cdot}13$$

Top Inter Bay.

The end loads in the spars are 5400 lbs. and 2894 lbs. for front and rear respectively.

$$\text{Front spar } P_{c_F} = \frac{\pi^2 \times 1{\cdot}6 \times 10^6 \times {\cdot}213}{25{\cdot}5^2} = 4140 \text{ lbs.}$$

$$\text{Rear spar } P_{c_R} = \frac{\pi^2 \times 1{\cdot}6 \times 10^6 \times {\cdot}24}{25{\cdot}5^2} = 4680 \text{ lbs.}$$

$$\frac{P_{c_F} + P_{c_R}}{P_F + P_R} = \frac{4140 + 4680}{5400 + 2894} = 1{\cdot}065$$

Top Centre Bay.

The end loads in the spars are 6176 lbs. and 1269 lbs. front and rear respectively.

$$\text{Front spar } P_{c_F} = \frac{\pi^2 \times 1{\cdot}6 \times 10^6 \times {\cdot}213}{24^2} = 5880 \text{ lbs.}$$

$$\text{Rear spar } P_{c_R} = \frac{\pi^2 \times 1{\cdot}6 \times 10^6 \times {\cdot}24}{24^2} = 6620 \text{ lbs.}$$

$$\frac{P_{c_F} + P_{c_R}}{P_F + P_R} = \frac{5880 + 6620}{6176 + 1269} = 1{\cdot}68$$

Bottom Outer Bay.

The end loads in spars are – 174 lbs. and 522 lbs., and therefore the spars are up to strength by inspection.

Bottom Inner Bay.

The end loads in the spars are – 2650 lbs. and 533 lbs. front and rear respectively, so that the spars are strong enough by inspection.

Case 2. Flying with 5 Times Load, C.P. at ·5 of Chord.

Top Outer Bay.

The end loads in spars are 392 lbs. and 2935 lbs. front and rear respectively.

P_{c_F} and P_{c_R} = 3120 lbs. and 3520 lbs. front and rear respectively.

$$\frac{P_{c_F}+P_{c_R}}{P_F+P_R} = \frac{3120+3520}{392+2935} = 2\cdot0$$

Top Inter Bay.

The end loads in spars are 1100 lbs. and 6246 lbs. front and rear respectively.

P_{c_F} and P_{c_R} = 5220 lbs. and 5900 lbs. respectively.

$$\frac{P_{c_F}+P_{c_R}}{P_F+P_R} = \frac{5220+5900}{1100+6246} = 1\cdot510$$

Top Centre Bay.

The end load in spars are 1223 lbs. and 6079 lbs. front and rear respectively.

P_{c_F} and P_{c_R} = 5880 lbs. and 6620 lbs. respectively.

$$\frac{P_{c_F}+P_{c_R}}{P_F+P_R} = \frac{5880+6620}{1223+6079} = 1\cdot71$$

Bottom Outer Bay.

The end loads in spars are – 292 lbs. and 877 lbs. front and rear, so that spars are all right by inspection.

Bottom Inner Bay.

The end loads in spars are – 2438 lbs. and 814 lbs. front and rear respectively, hence spars are strong enough by inspection.

CHAPTER VIII

FUSELAGE

The main function of the body or fuselage is to provide accommodation for cargo, pilot, passengers, flying instruments and a reliable bearing for the power unit. This also forms a means of connection between various other components on the aeroplane, such as the wings and tail unit.

It is one of those absolutely vital components in which the failure of any one member would be dangerous if not absolutely fatal to the whole machine, and therefore calls for the utmost care in the estimation of the stresses in every member.

No one case of loading may be assumed to be sufficient, without a consideration of the several other cases which are likely to affect the structure to any extent.

Of the several forms of construction in general use, the fuselage under consideration is of the girder type, and of rectangular cross section. The four longerons and all the cross struts are of wood; spruce and ash being the two kinds of timber used. Generally the engine bearers, bottom longerons for front portion of fuselage, and several of the cross struts are of ash, while the remainder are usually of spruce. The struts divide the frame into bays which are braced by means of swaged round wires and by which adjustment can be made. Cross bracing is inserted in as many bays as possible, in order to make the structure very rigid.

It is advisable to begin the calculations by drawing up a detailed list of all the components of the machine, together with the weight of each. This has been done in Table 12. It is also essential to know the position of the centre of gravity of the whole aeroplane, and for this purpose a side elevation is drawn (Fig. 42) indicating the positions of the centre of gravity of each separate component of the machine.

Experience will help in making good estimates of the positions of these centres of gravity (the centre of gravity of the whole machine, of course, cannot be verified until the machine is actually constructed) although many components, such as propeller, petrol, oil, radiator, cargo, etc., etc., present little or no difficulty.

Having fixed the positions of the centres of gravity of the

TABLE 12

List of Components.	Weight.
Wings (area 332½ sq. ft.), including struts and wires, wing struts 29 lbs., and wires 15 lbs.	330 lbs.
Petrol at ·6 pt. per B.H.P. hour, 3½ hrs. (say 40 galls. tank, etc., 40 lbs.)	330 ,,
Oil, 3½ galls. + tank and bearers (·05 pints per B.H.P. hour, 3½ hours)	38 ,,
Engine, 150 H.P. (French ungeared Hispano-Suiza) dry	440 ,,
Piping, complete for water, petrol, oil, etc.	24 ,,
Water in radiator, tank, pipes, etc.	70 ,,
Radiator and shutter	50 ,,
Engine accessories	12 ,,
Landing gear (wheels, 40 lbs.), Palmer 700 × 100	87 ,,
Propeller, 36·0; Boss, 17·0	53 ,,
Frame (front, 95 lbs., rear, 25 lbs.)	120 ,,
Tail Plane and elevators (40 sq. ft.)	40 ,,
Rear skid	6 ,,
Fins (6 sq. ft.)	7 ,,
Rudder (9 sq. ft.)	7 ,,
Pilot (complete with flying kit)	170 ,,
Controls in body	22 ,,
Fairing, cowling (front, 50 lbs.; rear, 20 lbs.)	70 ,,
Flying instruments	19 ,,
Seat, upholstered, etc.	15 ,,
Exhaust pipes	20 ,,
Cargo	400 ,,
Total Weight	2330 lbs.

Total load, 2330 lbs.
Wing area, 332½ sq. ft.
Load per $(\text{foot})^2 = 7{\cdot}0$ lbs.

components, the position of the centre of gravity of the whole machine is found by taking moments about two datum lines, the one horizontal and the other vertical. This has been done in Table 13 and Table 14 respectively, where the distance scale in each case is full size, the ordinates of the point so found being 74·6 in. horizontally and 60·3 in. vertically. This point, when obtained, should lie as near as possible on the thrust line of the propeller, and approximately at about ·28 of the mean chord of the wing surface, or to the satisfaction of the designer.

The loads due to the separate components must now be distributed over the joints of the fuselage, and, in so doing, attention must be paid to such components as front and rear fairing, front and rear fuselage, and other components whose loads are distributed over several joints. The centre of the distributed loads on the several joints (due to a component),

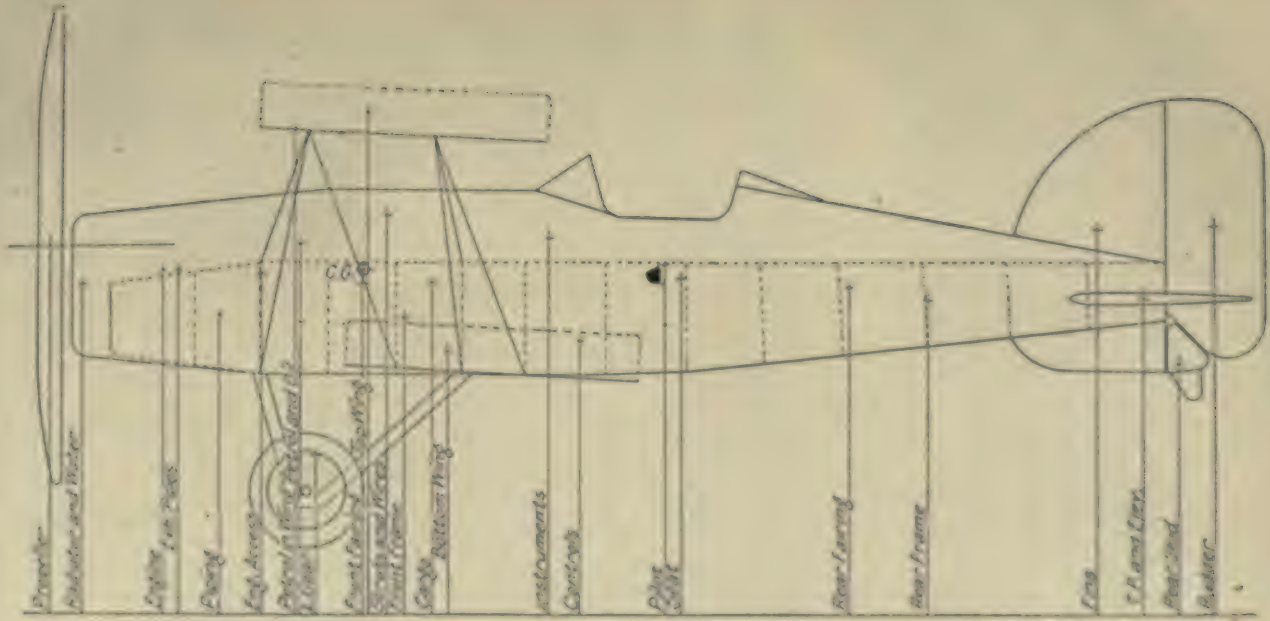


FIG. 42.—SIDE ELEVATION SHOWING C.G.'s OF COMPONENTS

must coincide as far as possible with the centre of gravity of the component already fixed, otherwise there will be a decided lack of balance, which will be troublesome when the reciprocal load diagrams are drawn.

It should also be noticed that in the case of many components the load is imposed on the upper or lower joints of the fuselage, but not both.

For example, the load due to wings is distributed over four different joints on each side of frame, partly on two upper joints, and partly on two lower joints, whereas the landing gear is wholly taken on two lower joints of the fuselage.

TABLE 13

Horizontal Position of C.G. Moments About Vertical Line Through Propeller

Petrol and tank (body), 32 galls.		290	×	57·0	=	16500
Oil and tank (body), 3½ galls.		38	×	57·0	=	2160
Propeller and boss.		53	×	0	=	0
Engine		440	×	27·0	=	11900
Water in radiator, tank, pipes,		70	×	7·5	=	530
Radiator and shutter		50	×	7·5	=	380
Landing gear		87	×	63·0	=	5480
Front frame		95	×	84·0	=	7980
Rear frame		25	×	208	=	5200
Tail plane and elevators		40	×	261	=	10440
Rear skid		6	×	269	=	1610
Fins, top and bottom		7	×	250	=	1750
Rudder		7	×	277	=	1940
Pilot		170	×	146	=	24800
Controls in body		22	×	126	=	2770
Front fairing		50	×	74	=	3700
Rear fairing		20	×	190	=	3800
Flying instruments		19	×	120	=	2280
Seat upholstered, etc.		15	×	150	=	2250
Exhaust pipes		20	×	30	=	600
Cargo		400	×	90	=	36000
Top wing (172·5 sq. ft.)	330	150	×	75	=	11250
Bottom wing (160 sq. ft.)		145	×	95	=	13780
Struts and wires		35	×	80	=	2800
Petrol in wing and tank		40	×	56	=	2240
Piping, complete		24	×	40	=	960
Engine Accessories		12	×	50	=	600
Total moment						173700

$$x \text{ horizontally} = \frac{173700}{2330} = 74{\cdot}6 \text{ in. from centre line of propeller.}$$

Distance scale, full size.

TABLE 14

VERTICAL POSITION OF C.G. MOMENTS ABOUT HORIZONTAL LINE

Petrol in body	290	×	66·4	=	19260
Oil in tank	38	×	48	=	1830
Propeller and boss	53	×	61·0	=	3230
Engine	440	×	61·0	=	26840
Water in radiator, tank, etc.	120	×	57	=	6840
Landing gear	87	×	17	=	1480
Front frame	95	×	49	=	4650
Rear frame	25	×	52	=	1300
Tail plane and elevators	40	×	54·5	=	2180
Rear skid	6	×	38	=	230
Fins, top and bottom	7	×	70	=	490
Rudder	7	×	72	=	500
Pilot	170	×	63	=	10700
Controls in body	22	×	43·5	=	960
Front fairing	50	×	57·5	=	2880
Rear fairing	20	×	56	=	1120
Flying instruments	19	×	68	=	1290
Seat upholstered	15	×	59	=	890
Exhaust pipes	20	×	61	=	1220
Cargo	400	×	58	=	23200
Top wing	150	×	100	=	15000
Bottom wing	145	×	43	=	6230
Struts and wires	35	×	73	=	2560
Petrol in wing	40	×	97	=	3880
Piping, complete	24	×	50	=	1200
Engine accessories	12	×	60	=	720
Total moment					140680

Total weight = 2330 lbs.

$\bar{y}$ vertically $= \frac{140680}{2330} = 60{\cdot}3$ in. from datum line.

Distance scale, full size.

This distribution is more tedious than difficult, and the methodical man will adopt his own procedure in the matter, but it is essential that some care should be exercised in order that the work may be easily and quickly checked at any stage.

The method adopted here is suggested as being quite satisfactory without undue and exhaustive detail, and it will be found that practically everything essential can be easily gathered from Table 15.

Every component, and consequently every load on the fuselage, is assumed to be symmetrical about the longitudinal axis of the machine. This means that the two side panels of the fuselage take half the full load of the whole aeroplane. The loads shown in Tables 15 and 16 represent 1 × load on the two sides of the fuselage, hence the loads at the joints must be halved and multiplied by the full factor required.

In the landing cases the whole of the top wing load, and the load due to the portion of the bottom wing from the tip to the centre of inner bay is transferred through the body struts to the top of the frame.

The remainder is taken by the two points of attachment of the bottom wing to the fuselage.

Since the rudder and skid are overhung loads at the rear of the machine, it is assumed that these are equivalent to a load at the fuselage, together with a horizontal couple.

This is quite allowable because of the way in which the rudder is fixed to the fuselage, but it is not so true in the case of the propeller and boss at the front of the machine.

Having tabulated the loads in Table 15, it is convenient to make a diagram of the frame and insert the loads according to their points of application, using judgment in placing above or below or in dividing between the two.

The way in which this has been done can easily be traced from the diagram in Fig. 43. (Inset).

TABLE 15

DISTRIBUTION OF LOADS ALONG THE FUSELAGE

Name of component.	Code letter on accompanying. diagram. Fig. 43.	Total Weight of component lbs.	Loads imposed on the joints due to each component.
Petrol in body .	*a*	290	Joint 3, 145 lbs. Joint 4, 145 lbs.
Oil, etc. . . .	*b*	38	Joint 3, 19 lbs. Joint 4, 19 lbs.
Propeller and boss	*c*	53	Joint 1, 53 lbs. With horizontal forces, 44 lbs. top and bottom of fuselage
Engine . . .	*d*	440	Joint 1, 180 lbs. Joint 3, 20 lbs. ,, 2, 240 ,,
Radiator, shutter, Radiator, water	*e*	70 50	Joint 1, 120 lbs. and 46·6 lbs. at top and bottom horizontal couple
Landing gear . .	*f*	87	Joint 3, 63 lbs. Joint 6, 24 lbs.
Front fairing . .	*g*	50	Joint 1, 7 lbs. Joint 6, 5 lbs. ,, 2, 7 ,, ,, 7, 5 ,, ,, 3, 6 ,, ,, 8, 5 ,, ,, 4, 6 ,, ,, 9, 4 ,, ,, 5, 5 ,,
Rear fairing . .	*h*	20	Joint 9, 5 lbs. Joint 13, 2½ lbs. ,, 10, 4 ,, ,, 14, 1 ,, ,, 11, 4 ,, ,, 15, 1 ,, ,, 12, 2½ ,,

TABLE 15 (*continued*)

Name of component.	Code letter on accompanying diagram. Fig. 43	Total Weight of component lbs.	Loads imposed on the joints due to each component.
Tail plane & elevator	*j*	40	Joint 14, 9·3 lbs. Joint 15, 30·7 lbs.
Rear skid . . .	*k*	6	Joint 15, 6 lbs and horizontal couple, 1·04 lbs. top and bottom
Fins	*l*	7	Joint 13, 1 lb. Joint 15, 2 lbs „ 14, 4 „
Rudder	*m*	7	Joint 15, 7 lbs. and 5·4 lbs. horizontal couple
Exhaust pipes .	*n*	20	Joint 1, 6 lbs. Joint 3, 2 lbs. „ 2, 12 „
Pilot . . .	*o*	170	Joint 8, 64 lbs. Joint 10, 16 lbs. „ 9, 90 „
Controls . .	*p*	22	Joint 7, 7 lbs. Joint 8, 15 lbs.
Flying Instruments	*q*	19	Joint 7, 12·2 lbs. Joint 8, 6·8 lbs.
Seat . . .	*r*	15	Joint 8, 3 lbs. Joint 10, 2 lbs. „ 9, 10 „
Cargo . . .	*s*	400	Joint 4, 69 lbs. Joint 6, 132 lbs. „ 5, 137 „ „ 7, 62 „
Piping complete .	*t*	24	Joint 2, 15.1 lbs. Joint 3, 8·9 lbs.
Engine accessories	*u*	12	Joint 3, 12 lbs.
Top wing . .	*v*	150	Joint 3, 72 lbs. Joint 6, 78 lbs.
Bottom wing .	*w*	145	Joint 3, 10 lbs. Joint 6, 110 lbs. „ 5, 15 „ „ 7, 10 „
Struts and wires .	*x*	35	Joint 3, 13·5 lbs. Joint 6, 21·5 lbs.
Petrol in wing .	*y*	40	Joint 3, 25 lbs. Joint 6, 5 lbs.
Front fuselage .	*z*	95	Joint 1, 10 lbs. Joint 5, 10 lbs „ 2, 10 lbs. „ 6, 12 „ „ 3, 9 „ „ 7, 13 „ „ 4, 11 „ „ 8, 12 „ „ 9, 8 „
Rear fuselage .	*A*	25	Joint 9, 4 lbs. Joint 13, 3½ lbs. „ 10, 3½ „ „ 14, 3 „ „ 11, 3½ „ „ 15, 4 „ „ 12, 3½ „

As a check on the work the load should total up to the weight of the complete machine. Similarly, by taking moments about the centre line of propeller, the total moment should compare favourably with that in Table 13.

The factor of safety required in landing is at the present time more or less standard for all machines, and is fixed at 5, although for fairly large aeroplanes 4 is permissible.

When the machine is flying the factors correspond to those used for the main planes.

It is now required to set out a side elevation of the fuselage diagrammatically to some scale and insert the loads on one side of fuselage, including a factor of 5; *i.e.*, the loads at the joints in Fig. 43 (which are loads on the two sides of the fuselage) must be multiplied by 2½ in order to give a factor of 5 times load on each side panel of the fuselage.

Case 1, Landing on Wheels at 5 Times Load

This is the geral condition of normal landing when the whole shock is taken through the wheels and landing chassis to the fuselage, some amount of absorption having taken place at the tyres and rubber, or other absorbers. It is assumed that the direction of the reaction due to this case of landing is along the line joining the point of contact with the ground and the centre line of gravity of the whole machine, and consequently the direction of the loads will be parallel to this line. The work of finding the loads in the various members of the structure is done graphically in the ordinary way by lettering the position diagram according to Bow's system of notation.

It serves as a good check and often avoids considerable delay if it is remembered that the external forces on the structure are themselves in perfect balance, hence it is advisable first to draw the reciprocal force diagram for the external forces, and see that this diagram completely closes. The reciprocal diagram may now be completed and the loads found in the various members of the frame structure. It will be seen in Fig. 44 (inset), that it has been found convenient to do this in two diagrams drawn to different scales, the larger diagram representing the loads on the front portion, while the smaller diagram represents the load on the rear portion of the fuselage.

From these diagrams the loads are measured and inserted on the particular members in the diagram.

The fuselage may be considered supported at the front and rear landing leg attachments, of which the portions outside these two points are cantilevers. Every joint should be checked over to see that the right wires are in operation, otherwise they may be found to be in compression, which is impossible.

Case 2, Landing on Wheels and Rear Skid at 5 Times Load

The above case is the one in which the shock is divided proportionately between the wheels and the rear skid, both being assumed to touch the ground simultaneously, the greater proportion of load being taken by the wheels, since they are nearer to the centre of gravity of the whole machine. It represents a very severe case for the rear portion, and induces loads in many of the members which determine their design. It is a form of landing rather difficult to make, and is generally considered to be the correct method.

To find the load on the wheels and skid respectively, drop a perpendicular on to a line drawn tangent to the wheel, and touching the skid in its partially compressed position. The load on the skid and wheels can then be found by proportion.

In this example these proportions are 213 and 22, and since the weight of the whole machine is 2330 lbs., then

$$\frac{2330 \times 22}{213} = 240 \text{ lbs. at the skid,}$$

the load at the wheels being—

$$2330 - 240 = 2090 \text{ lbs.}$$

These loads represent 1 × loading on both sides of the machine, hence for a factor of 5 times load, the reactions on one side of the frame are—

$$2\tfrac{1}{2} \times 2090 = 5220 \text{ lbs. at the wheel.}$$

$$2\tfrac{1}{2} \times 240 = 600 \text{ lbs. at the skid.}$$

The loads at the several joints may now be inserted according to the distribution already made, taking care that the factor is 5 times load at every joint.

All these dead load forces have a direction perpendicular to the ground line, and loads in the members of the structure may be found either by calculation, or by drawing out a stress diagram graphically. In this case, as in Case 1, the graphical method has been employed, and the diagrams will be found in Fig. 45. (Inset.)

The magnitudes are measured from this reciprocal diagram and inserted on the particular members.

Checks on the work should be made wherever possible, so that any errors may be eliminated.

It will be seen that the fuselage in this case may be likened to a structure supported at three points (*i.e.*, skid, front and rear landing legs), the front portion being overhung.

This fact will help in deciding which wires are operative, when the force diagram is being drawn.

CHAPTER IX

FUSELAGE (*Continued*)

Case 3, Flying with C.P. at ·5 of the Chord 5 Times Load

This is the case for an aeroplane flying with a small angle of incidence (about 3 degrees) the engine working at open throttle and the machine fully loaded.

The forces operating on the machine are—

(1) Thrust of the propeller.

(2) Normal weight of the aeroplane (*i.e.*, total weight minus the weight of the wing structure) distributed over the joints of the fuselage.

(3) Drag forces.

(4) Lift forces in the lift wires and body struts.

(5) Possible tail load, which is generally a down load on the tail in order to maintain the machine in this flying position.

These forces operate simultaneously and should be in exact balance for steady flight.

The thrust of the propeller may be taken as being equal to the total drag forces. Since the wings are air borne, the weight of the wing structure cannot stress the fuselage in any way whatever.

The tail load is, of course, a variable quantity and must be found for the particular aeroplane in the case under consideration by taking moments of all the operating forces during flight, about some convenient point, such as the centre of gravity.

The load factor is again 5, hence the normal load on one side of the machine

$$= 5 \times \frac{\text{normal load}}{2}$$

$$= 5 \times \frac{2000}{2} = 5000 \text{ lbs.}$$

In proportion to the areas and efficiency the loads on the top and bottom planes are 2870 lbs. and 2130 lbs. respectively, and taking drags to be one-seventh of the lifts, then drag force on top wing = 410 lbs., and on bottom wing = 315 lbs.

The lift load on the top wing is partly transferred to the fuselage by the lift wires, and partly through the body struts, in the proportions 2100 lbs. and 770 lbs. respectively, according to areas.

Similarly, the lift load on the bottom wing is taken on the lift wires, and at the fuselage attachments in the proportions 1780 lbs. and 350 lbs. respectively.

The total load to be taken by the lift wires is therefore 2100 + 1780 = 3880 lbs., and this is to be divided proportionately between the front and rear main lift wires.

Of this total load, 1780 lbs. is transferred from bottom plane to top plane, and incidentally is moved forward an amount depending on the stagger, which introduces a moment

$$1780 \times 24 = 42700 \text{ lbs. ins.}$$

It is known that the position of the centre of pressure is at ·5 of the chord, consequently the vertical components of the lift wire loads may be found by proportion, and will be found to be 1260 lbs. and 2620 lbs. at front and rear lift wire attachments respectively.

The lift loads at the top front and rear body strut joints are found by proportion to be 190 lbs. and 580 lbs. respectively, whilst the lift loads at the bottom front and rear wing attachment joints are 94 lbs. and 256 lbs. respectively.

In order to find the tail load, take moments about the centre of gravity of the machine, viz:—

Front lift wire	1260	×	25	ins.	=	+	30500	lbs. ins
Rear lift wire	2620	×	24	,,	=	–	62800	,,
Top front body-strut joint	190	×	14·6	,,	=	+	2770	,,
Top rear body-strut joint	580	×	15·4	,,	=	–	9020	,,
Bottom front attachment joint	94	×	9·0	,,	=	–	850	,,
Bottom rear attachment joint	256	×	39	,,	=	–	10000	,,
Moment of load transferred from bottom to top wing	1780	×	24	,,	=	–	42700	,,
Top drag force	410	×	31	,,	=	+	12700	,,
Bottom drag force	315	×	25	,,	=	–	7900	,,
Tail load	L	×	189	,,	=	+	$189L$	,,
Normal load	5000	×	0	,,	=		0	,,
Thrust	725	×	0	,,	=		0	,,

(+ and – signs denote stalling and nose diving moments respectively.)

Summing up the moments, there is found to be an unbalanced moment which requires a down load on the tail of 460 lbs. This is, of course, only half the actual load, as we are considering one side panel of the fuselage.

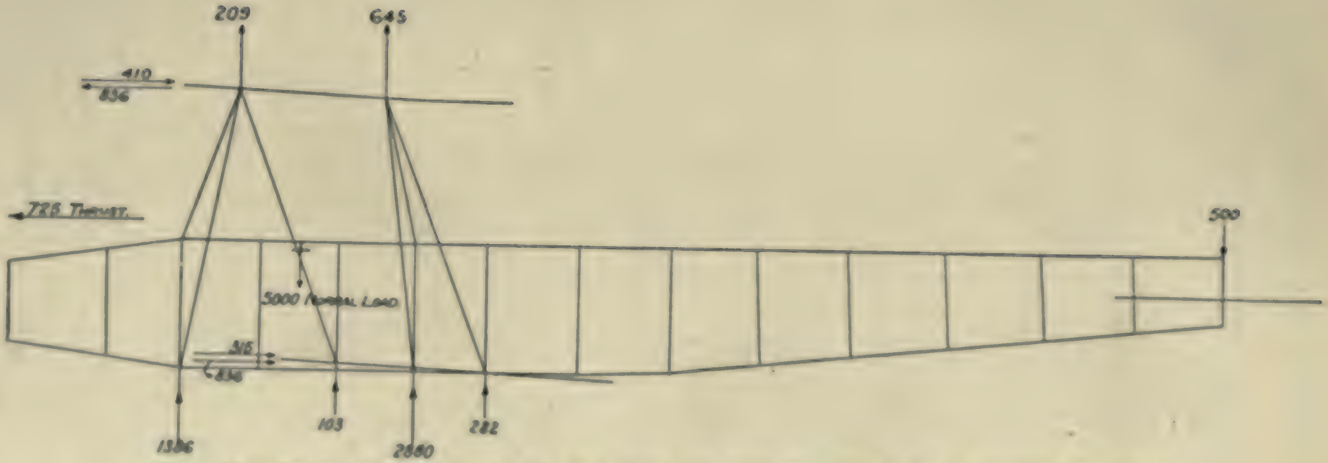


FIG. 46.—DIAGRAM OF LOADS ON ONE SIDE OF THE FUSELAGE FOR FLYING WITH C.P. AT ·5 OF CHORD, 5 TIMES LOAD

This down load on the tail will in turn necessitate an increase of all the lift loads roughly in the proportion of $\frac{5500}{5000}$.

Having done this, moments are again taken about the C.G. of the machine and the more correct tail load determined.

This modified tail load is found to be 500 lbs. (*i.e.*, 4·0 lbs. per ft^2. at normal load).

The loads at the several joints are as shown in the Fig. 46.

Stagger Loads

The load transferred from bottom to top is now 1980 lbs., and since the stagger is 24 ins., and the gap of the planes 56 ins., two equal and opposite forces are introduced, the one on the top plane being forward and of magnitude and

$$= \frac{1980 \times 24}{56} = 836 \text{ lbs.}$$

Thrust of the propeller is equal to the resultant drag forces = 725 lbs., and is assumed to have no moment about the C.G.

The three conditions of equilibrium have now been satisfied, namely—

(1) Algebraic sum of the vertical forces is zero.

(2) Algebraic sum of the horizontal forces is zero.

(3) Algebraic sum of the moments of all the forces is zero, which means, that if the work of finding the loads in the members of the fuselage is done graphically, then the force diagram will close, excepting possible accumulated errors in drawing.

If it is decided to do the work by calculation the usual method, known as the "method of sections," is applied. This method will be found described in most books on Graphics or Structural Engineering.

In the present case the work has been done graphically, and Fig. 47 shows a scale diagram of the fuselage with all the flying loads applied thereto.

It will be noticed that the external loads due to the dead weight of the machine are practically the same as in Cases 1 and 2, Figs. 44 and 45; the notable exceptions being that the loads at joints 3, 5, 6, and 7 (due to the weight of the wing structure) have been removed. Thus at joint 3 the load is

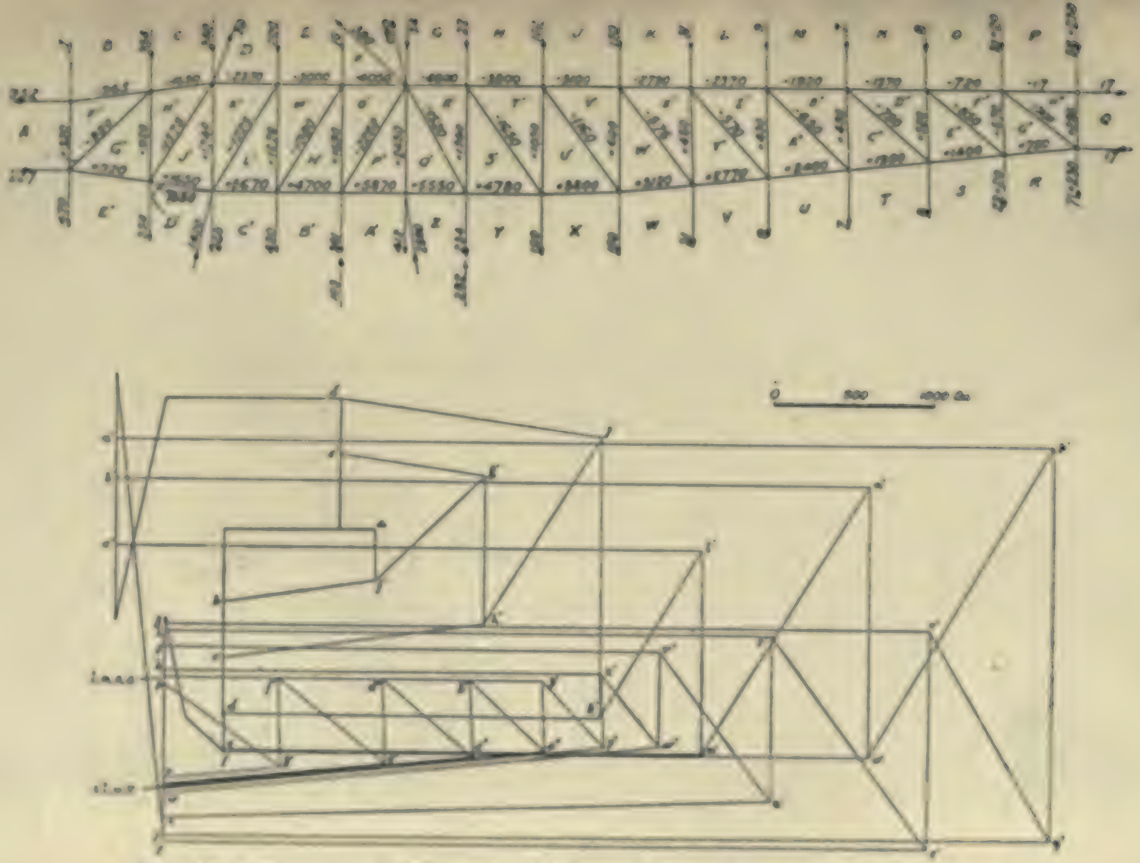


FIG. 47.—RECIPROCAL DIAGRAM. FLYING WITH C.P. AT ·5 OF THE CHORD, 5 TIMES LOAD

reduced by 240 lbs., at joint 5 by 37·5 lbs., at joint 6 by 522 lbs., and at joint 7 by 25 lbs.

The loads in the body struts and bracing are obtained from the diagram in Fig. 48, where the applied loads are the lifts, and the resultant drag.

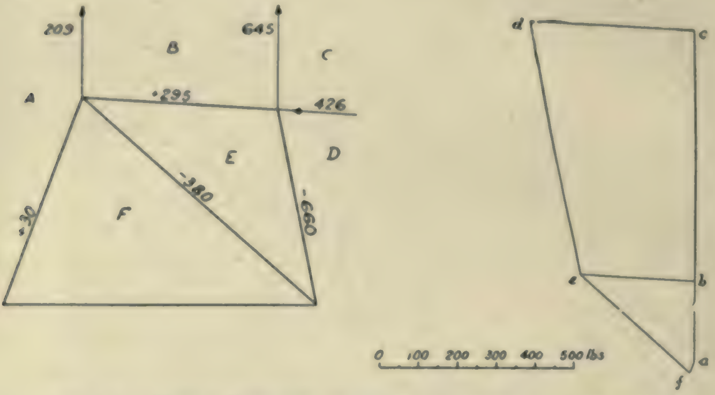


FIG. 48.—DRAG FORCES IN BODY STRUTS AND BRACING. FLYING, CASE 3

When the force diagram is completed the loads in the various members are measured from it, and inserted on the diagram. (*See* Fig. 47.)

Sometimes it is possible to sustain an up load on the tail, but when it does occur it is generally small.

This condition for an upload on the tail is that in which the centre of gravity of the aeroplane is aft of the centre of pressure of the lifting surfaces. It is usual, however, to assume that it is possible to have an up load on the tail load when considering the strength of that unit, and so the tail unit must be checked accordingly.

Case 4, Flattening Out at 5 Times Load + 1 Times Inertia Load

This is probably the most severe of the cases it is necessary to consider for the fuselage, and in order to get an idea of the forces operating, the aeroplane may be imagined to be just recovering, or about to recover, from a very steep dive under a very heavy tail load of about 30 lbs. per square foot.

This tail load has the effect of flattening out the steepness of the dive, and it is just when the tail load is imposed by the pilot operating the elevators that the fuselage will be stressed most severely. Not only are there loads due to the dead weight of the machine, but there are also inertia forces due to the angular acceleration produced by the heavy tail load. When the machine is diving it is quite probable that the centre of pressure is well back, but when flattening out it begins to move forward. In working out this case, it is assumed that the machine has so far recovered that the total lift on the wings is equal to the sum of the normal load of the aeroplane, added to the tail load.

The factor of loading employed is the same as in the other cases (*i.e.*, 5 times load), but with the addition of 1 times inertia loads, which is quite reasonable.

When flattening out the operating forces are—

(1) Thrust of the propeller.

(2) Normal weight of the machine.

(3) Drag forces.

(4) Lift forces in lift wires, body struts, and bracing.

(5) Down load on the tail.

(6) Inertial forces on the whole machine; and since they operate simultaneously they are in exact balance during the period of flattening out.

As in Case 3, the thrust of the propeller is assumed equal to the total drag forces, and the drag forces are taken as one-seventh of the normal lift.

$$\text{Normal load} = 5 \times \frac{2000}{2} = 5000 \text{ lbs.}$$

on one side of fuselage, and this is divided into 2870 lbs. lift on the top plane, and 2130 lbs. lift on the bottom plane.

The drag force on the top plane is therefore 410 lbs., and on the bottom plane 315 lbs. Maximum down load on the tail is obtained from Fig. 49 and amounts to 1500 lbs.; or 750 lbs. on each half of the tail.

Inertia Forces.

Since the load on the tail plane is known, its moment about the cross wind axis of the machine may be calculated. This moment gives the aeroplane an acceleration of amount depending upon the moment of inertia of the machine about that axis.

For the purpose of estimating the moment of inertia of the

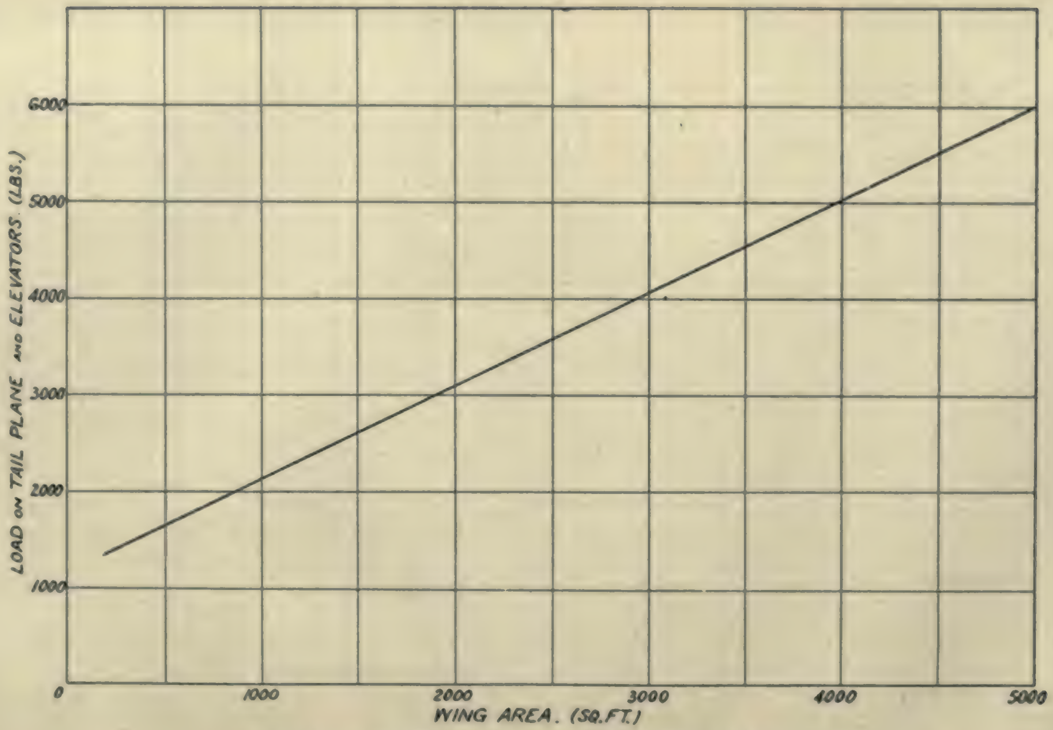


FIG. 49.—TAIL LOAD FOR A GIVEN WING AREA

whole machine a table is prepared as shown in Table 16, in which the moment of inertia of each component is calculated. If a component of weight w lbs., which has no moment of inertia about its own axis, is situated at a radius r ft. from the C.G. of the aeroplane, then $I = \frac{w}{g} r^2$ where I is the moment of inertia in engineers' units, about the cross wind axis.

Table 16 is a methodical means of obtaining the I of each component on this assumption, and the sum of these values gives the I of the whole machine.

It is not strictly accurate, however, to make this assumption in the case of such components as engine, wings, tail plane, etc., which have moments of inertia about their own axes. Consequently an increase of 7% is made to the I of the whole machine to make allowance for this fact.

From the tabulation the I of the machine is 1436, and when increased by 7% = 1532 units.

The distance from the C.P. on the tail to the C.G. of the machine is 15·1 ft., and since the tail load is 1500 lbs., then the applied torque is—

$$1500 \times 15{\cdot}1 = 22700 \text{ lbs. ft.}$$

Now Ia = torque, where a is the angular acceleration of the machine about its C.G., hence these two quantities must be equated in order to determine the angular acceleration.

Thus—

$$1532\, a = 1500 \times 15{\cdot}1$$

$$\therefore\ a = \frac{1500 \times 15{\cdot}1}{1532} = 14{\cdot}8 \text{ radians per sec., per sec.}$$

It is generally assumed, however, that approximately one-half of this angular acceleration is damped out by the wings, in consequence of which the actual angular acceleration of the machine is 7·4 radians per sec., per sec.

Each component exerts an inertia force on the fuselage, depending upon its radius and its mass (*i.e.*, inertia force $= \frac{w}{g} r\, a$) and operates in a direction perpendicular to the line joining the C.G. of the component to the C.G. of the whole machine, and tends to oppose the turning effect of the tail load.

The magnitudes of these inertia forces are tabulated in

Table 16, and the vertical components of these forces in Table 17, horizontal components being generally neglected.

TABLE 16

DISTANCE SCALES—FULL SIZE

Component.	Weight W lbs.	$\frac{W}{g}$	Rad. (ins.) from CG	Rad. R ft.	R^2 (ft.)	$\frac{W}{g} R = \frac{I}{R}$	$\frac{W}{g} R^2 = I$
Petrol and tank	290	9·0	18·6	1·55	2·4	13·95	21·60
Oil and tank	38	1·18	22·0	1·83	3·35	2·16	3·95
Propeller and boss	53	1·646	75·0	6·25	39·1	10·30	64·30
Engine	440	13·68	48·0	4·00	16·0	54·75	219·00
Water and radiator, shutter, etc.	120	3·73	67·5	5·62	31·65	20·90	118·00
Landing gear	87	2·7	45·0	3·75	14·05	10·10	38·00
Front frame	95	2·95	15·0	1·25	1·561	3·69	4·60
Rear frame	25	·777	134·0	11·17	124·7	8·67	96·80
Tail plane and elevators	40	1·24	186·5	15·54	242·0	19·30	300·00
Rear skid	6	·186	197·0	16·42	270·0	3·06	50·30
Fins	7	·214	175·0	14·58	212·2	3·11	45·30
Rudder	7	·214	203·0	16·92	286·2	3·61	61·20
Pilot	170	5·28	71·6	5·97	35·7	31·50	189·00
Controls in body	22	·684	54·5	4·54	20·6	3·06	13·90
Front fairing	50	1·55	3·0	·25	·063	·31	0·08
Rear fairing	20	·62	115·5	9·62	93·0	5·96	57·70
Flying instruments	19	·59	46·5	3·87	15·0	2·28	8·85
Seat, etc.	15	·465	75·5	6·29	39·5	2·92	18·40
Exhaust pipes	20	·62	45·0	3·75	14·05	2·33	8·70
Cargo	400	12·41	14·5	1·31	1·462	15·00	18·15
Top wing	150	4·65	39·5	3·29	10·8	15·30	50·20
Bottom wing	145	4·5	27·5	2·29	5·25	10·30	23·60
Struts and wires	35	1·09	13·5	1·125	1·27	1·22	1·37
Petrol in wing	40	1·24	41·0	3·42	11·67	4·24	14·50
Piping complete	24	·745	36·2	3·02	9·15	2·25	6·80
Engine accessories	12	·37	25·0	2·08	4·34	·77	1·61

TOTAL I (uncorrected) 1436·

It will be observed that some of these forces are upward, denoted by negative signs, and the remainder are downward, denoted by positive signs, depending upon their position relative to the C.G. (*i.e.*, positive for forces in front of the C.G., and negative for forces behind C.G.).

These inertia forces are still to be distributed over the frame in exactly the same proportions as for the dead weights themselves. This has been done in Fig. 50 (inset), but it must be remembered that everything in that figure is for the two sides of the machine, and must be divided by two in order to correspond with the previous work.

TABLE 17

Component.	Vertical factor.	$\frac{W}{g} R a$ Inertia force (lbs.)	Vertical force. (lbs.)
Petrol and tank	·9	103	+ 93·0
Oil and tank	·8	16	+ 13·2
Propeller and boss	·95	76·5	+ 72·5
Engine	·99	405	+ 400·0
Water and radiator	·99	152	+ 150·0
Landing gear	·27	75	+ 20·2
Front frame	·63	27·2	− 17·2
Rear frame	·99	63·5	− 63·0
Tail plane, etc.	1·00	143	− 143·0
Rear skid	·99	22·6	− 22·4
Fins	1·00	22·9	− 22·9
Rudder	1·00	26·7	− 26·7
Pilot	1·00	234	− 234
Controls	·98	22·6	− 22
Front fairing	·25	2·4	+ 0·6
Rear fairing	1·00	44	− 44
Flying instruments	·98	16·9	− 16·5
Seat	1·00	21·7	− 21·7
Engine pipes	1·00	17·2	+ 17·2
Cargo	·99	111	− 110
Top wing	0	113	0
Bottom wing	·65	76·4	− 49·7
Struts and wires	·36	9·06	− 3·3
Petrol in wing	·45	31·5	+ 14·2
Piping	·98	16·2	+ 15·8
Engine accessories	1·00	5·6	+ 5·6

It is now possible to take preliminary moments about the C.G., in order to find the position of centre of pressure of the wings for this case—

Tail moment . . . . 750 × 181 = + 137000 lbs. ins.
Top drag moment . . . 410 × 31 = + 12700 ,,

Total stalling moment . . . . . + 149700 ,,

Normal load moment . . . . . . = zero
Inertia forces moment . . . . . . = − 57300 lbs. ins.
Thrust moment . . . . . . = zero
Bottom drag moment . . . . . . = − 7900 ,,

− 65200 ,,
Resultant = 149700 − 65200 = 84500 lbs. ins.
Lift moment about C.G. . . . . . = − 5750 × x

+ and − signs denote stalling and nose diving moments respectively.
Total lift force on the planes, per 1 side of machine = normal load + tail load
= 5000 + 750 = 5750 lbs.

Therefore the distance of C.P. behind the C.G. of the machine

$$= \frac{84500}{5750} = 14{\cdot}7 \text{ ins.}$$

From previous work the ratio of the lift L, on the top and L on bottom planes is 1 to ·767.

$$\therefore\ 1{\cdot}767L = 5750 \text{ lbs.}$$

from which lift on top plane = 3260 lbs.
and lift on bottom plane = 2490 lbs.

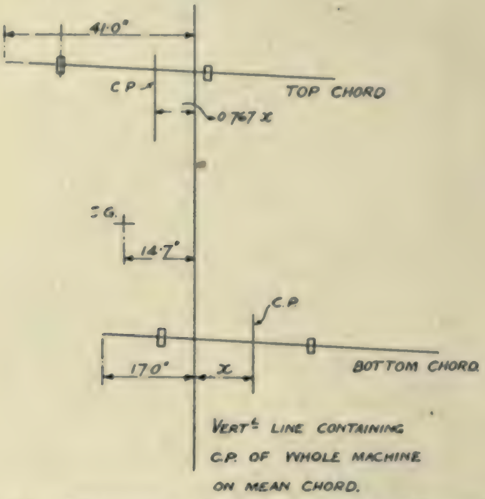


FIG. 51.—DIAGRAM TO FIND CENTRE OF PRESSURE

Of the load on the bottom plane, 390 lbs. is taken directly at the fuselage and 2100 lbs. is transferred to the top plane to be taken by the main lift wires.

The top plane load is partly taken on the body strut joints and the main lift wires in the proportions 900 lbs. and 2360 lbs. respectively. When the position of the C.P. is known relative to the chord, then these loads may be divided between front and rear.

In order to do this, draw the diagram as shown (Fig. 51) from the side elevation of the machine. Then

$$41 \text{ in.} - {\cdot}767x = 17 + x$$

Hence $x = 13{\cdot}6$ in., which corresponds to position of C.P. at ·445 × chord. The proportions of load are therefore—

Front body strut	= 350 lbs.
Rear body strut	= 550 „
Bottom front spar attachment to fuselege	= 150 „
Bottom rear spar attachment to fuselage	= 240 „
Vertical load at front lift wire attachment to fuselage	= 1790 „
Vertical load at rear lift wire attachment to fuselage	= 2670 lbs.

Stagger Loads

The load transferred from bottom plane to top plane introduces a nose-diving moment due to the stagger (24 ins.) of

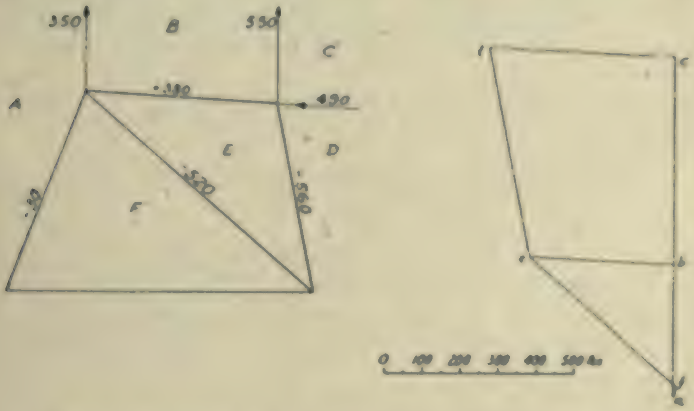


FIG. 53.—DRAG LOADS IN BODY STRUTS AND BRACING FLATTENING OUT. CASE 4

2100 × 24 = 50400 lbs. ins., and since the gap = 56 ins., a horizontal couple is produced on the wing structure, whose forces are 900 lbs.

As a check to see if the loads are in balance, take moments of all the forces about the C.G., from which it will be found that the resultant moment is comparatively very small.

Tail moment	750 × 181 =	+ 137000	lbs. ins.
Top drag	410 × 31 =	+ 12700	„
Inertia moment		− 57300	„
Thrust moment		zero	
Normal load moment		zero	
Bottom drag moment		− 7900	„

Front lift	1790 × 25	= +	44700	lbs. ins.
Rear lift	2670 × 24	= –	64100	,,
Front body strut . . .	350 × 14·6	= +	5110	,,
Rear body strut	550 × 15·4	= –	8470	,,
Bottom front attachment . .	150 × 9·0	= –	1350	,,
Bottom rear attachment . .	240 × 39	= –	9370	,,
Moment of load transferred from bottom to top		= –	50400	,,

which balance to within ¼%.

All the loads may now be transferred to a scale diagram and the forces in the members determined.

In this instance the loads in the members of the fuselage have been determined graphically, the diagrams for which are shown in Fig. 52. (Inset).

The true loads in the body struts and bracing are found graphically in Fig. 53 by applying the resultant drag force on the top wing, and the lifts at the two body strut joints.

CHAPTER X

FUSELAGE—(*Continued*)

TORSION OF FRONT PORTION OF FUSELAGE

This is due to the torque reaction of the engine, and is assumed to be equally distributed over the engine bearers. The loads in the various members due to engine torque will add to those already existing in normal flight and in flattening out.

If the normal horse-power given out by the engine is assumed 150 H.P., at a speed of 1400 revolutions per minute, then the engine torque may be calculated as follows—

$$T\,2\pi\,n = \text{H.P.} \times 33000$$

where T = engine torque in lbs. ft.

n = revolutions per minute

$$\text{then} \quad T = \frac{150 \times 33000}{2\pi\ 1400} = 565 \text{ lbs. ft.}$$

If now it is assumed that this torque be taken by the engine bearer bays as up load on one side and down load on the other, then, since the distance between the two sides of the machine is 29 ins., the up and down loads will be

$$\frac{565 \times 12}{29} = 234 \text{ lbs.}$$

The two engine bearers themselves, upon which the engine feet rest, are usually a less distance apart than the width of the fuselage. In the case of this engine the distance between the engine bearers is 13·2 in., so that the up load on one and the down load on the other engine bearer is

$$\frac{565 \times 12}{13{\cdot}2 \text{ in.}} = 512 \text{ lbs.}$$

having a shorter arm from the centre line of the machine.

It will therefore be realized that the difference between the force at the engine bearers themselves, and that at the side bays means that a lateral torque is introduced, which is usually taken by the ply bulkheads connecting the two engine bearers and the side bays.

In flight the whole engine torque is controlled and counterbalanced by the ailerons, which afford lateral stability to the machine.

This load of 234 lbs. is to be distributed over the three joints which support the engine bearers, hence each joint must take $\frac{234}{3} = 78$ lbs. It depends upon the direction of rotation of engine as to which side of the frame must withstand the down load, although both sides are usually made equally strong.

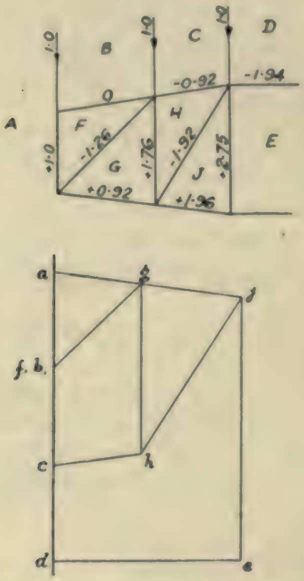


FIG. 54.—TORSION OF FRONT PORTION OF FUSELAGE

It will be seen that the up load will tend to relieve the frame of some of its ordinary flying loads, whereas the down load imposes additional loads.

This latter condition therefore constitutes the worst case of loading and is the one now considered.

Fig. 54 shows the front portion of the frame loaded equally at the three joints with unit load, and Table 18 shows the length factors obtained from the force diagram, from which actual loads in the several members are easily and quickly calculated. A factor of 2 has been allowed for this case, and is included in the loads in the last column of Table 18.

TABLE 18

Member.		Factor from force diagram under unit load.	Load in members due to 3 equal loads of 78 × 2 lbs.
Top	*BF*	—	—
Longeron	*CH*	·92	− 144
Side	*JE*	2·75	430
Struts	*GH*	1·76	275
	AF	1·00	156
Bottom	*AJ*	1·96	306
Longeron	*AG*	·92	144
Wires	*FG*	1·26	− 197
	HJ	1·92	− 300

TORSION OF REAR PORTION OF FUSELAGE

Owing to the fact that the portion of fin and rudder area above is often in excess of that below the fuselage, it will be conceived how that, due to side wind, a twisting effect is experienced by the rear portion of the fuselage.

The magnitude of this twisting effect depends upon the total side load, and also upon the distance of the centre of pressure of the combined fin and rudder from the centre line of the fuselage.

Simultaneously with this side load it is possible to experience a load on the tail plane and elevators, and it is usual to consider the rear portion of the fuselage for the combined cases.

The intensity of side load is assumed to be one-half of the maximum load intensity ever likely to be imposed upon the rudder and fins (*i.e.*, $\frac{15}{2} = 7{\cdot}5$ lbs. per square foot, and $\frac{10}{2} = 5$ lbs per square foot respectively. The maximum load ever likely to be imposed upon the tail unit has been taken as 1,500 lbs. in the flattening out case, hence the load in this case is 750 lbs. on the whole tail unit.

The skid is considered as being part of the rudder, and therefore takes a load in this case of 7·5 lbs. per square foot.

The combined area of top and bottom fins is 8 sq. ft., and the area of the rudder is 9 sq. ft., the area of the skid ·6 sq. ft.

Treating the several units separately and finding the moments of the side loads upon them about the centre line of the fuselage, we have—

Area of rudder . . . = 9 sq. ft.
Load on rudder . . . = 9 × 7·5 = 67·5 lbs.

Distance of centre of pressure of rudder from centre line of fuselage = 17 ins., hence the moment of load on rudder.

$$= 9 \times 7{\cdot}5 \times 17 = 1150 \text{ lbs. ins.}$$

Similarly for top and bottom fins, and also for skid—

Area of top fin . . . = 6 sq. ft.
Load on top fin . . . = 6 × 5 = 30 lbs.
Moment of this load . . = 30 × 23 = 690 lbs. ins.
Area of bottom fin . . . = 2 sq. ft.
Load on bottom fin . . = 5 × 2 = 10 lbs.
Moment of this load . . = 10 × – 12·5 = – 125 lbs. ins.
Area of skid . . . . = ·6 sq. ft.
Load on skid . . . . = ·6 × 7·5 = 4·5 lbs.
Moment of this load . . = 4·5 × – 16 = – 72 lbs. ins.
Net moment . . . . = 1150 + 690 – 125 – 72 = 1643 lbs. ins.
Total side load P_2 from port to starboard . . . = 67·5 + 30 + 10 + 4·5 = 112 lbs.

It will be observed from Fig. 55 that the fuselage, which usually tapers away in the rear, is assumed to maintain its rectangular shape for the purpose of estimating loads due to torque.

The load on the tail is assumed symmetrical about the fuselage, thus the load on one-half the tail is 375 lbs., which has no tendency to twist the structure.

In this case the rear portion of the fuselage is approximately square in cross section (Fig. 55), hence the arms of the two equivalent couples are equal. The forces F_1 and F_2 are, therefore, $F \times 14 \times 2 = 1643$, when $F = 58{\cdot}7$ lbs., for $d_1 = d_2 = 14$ ins.

If the rear portion of the frame is not square in cross section the arms of the couples are not equal, nor are the couple forces. In such a case the following consideration is applicable (*see* Fig. 55)—

The torque due to side load = T, then

$$T = F_1 \times d_1 + F_2 \times d_2 \quad . \quad . \quad . \quad . \qquad (a)$$

Furthermore, the ratio of F_1 to F_2 is equal to the ratio of the sines of the angles a_1 and a_2; *i.e.*—

$$\frac{F_1}{F_2} = \frac{\sin a_1}{\sin a_2} \quad . \quad . \quad . \quad . \quad . \quad . \qquad (b)$$

from which F_1 and F_2 may be calculated.

It is assumed that an up load of 750 lbs. on the tail unit may possibly occur instead of the down load already mentioned.

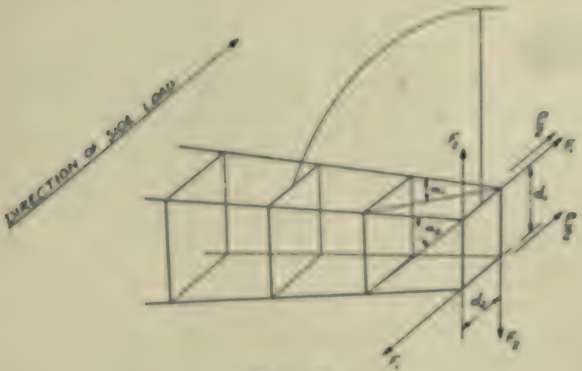


FIG. 55

The whole is therefore resolved into a consideration of the following, of which the up and down tail loads are alternative, and cannot occur simultaneously—

(1) Direct side load on top bays 56 lbs. port to starboard.
(2) Direct side load on bottom bays 56 lbs. port to starboard.
(3) Side load due to torque on top bays 58·7 lbs. port to starboard.
(4) Side load due to torque on bottom bays 58·7 lbs. starboard to port.
(5) Down load 375 lbs. on port side bays. Down load on tail.
(6) Down load 375 lbs. on starboard side bays. Down load on tail.
(7) Up load 58·7 lbs. on port side bays due to torque.
(8) Down load 58·7 lbs. on starboard side bays due to torque.
(9) Up load 375 lbs. on port side bays. Up load on tail.
(10) Up load 375 lbs. on starboard side bays. Up load on tail.

(5) and (6) cannot operate in conjunction with (9) and (10), but either may operate along with side load.
(1) and (3) may be combined (*i.e.*, 56 + 58·7 = 114·7 lbs. on top bays from port to starboard) . . . . . . (11)
(2) and (4) may be combined (*i.e.*, 56 − 58·7 = − 2·7 lbs. on bottom bays from port to starboard) . . . . . (12)
(5) and (7) may be combined in case of down load on tail—
375 − 58·7 = 316·3 lbs. down load on port side bays (13)
(6) and (8) may be combined in case of down load on tail—
375 + 58·7 = 433·7 lbs. down load on starboard sidebays (14)
(7) and (9) may be combined in case of up load on tail—
375 + 58·7 = 433·7 lbs. up load on port side bays . (15)
(8) and (10) may be combined in case of up load on tail—
375 − 58·7 = 316·3 lbs. up load on starboard side bays (16)

It is assumed that the whole of the side and tail load operates at the extreme end of the fuselage and two diagrams only need be drawn, one for the top and bottom bays, and one for the two side bays. This has been done in Figs. 56 and 57, in which unit load is placed at the end of the fuselage.

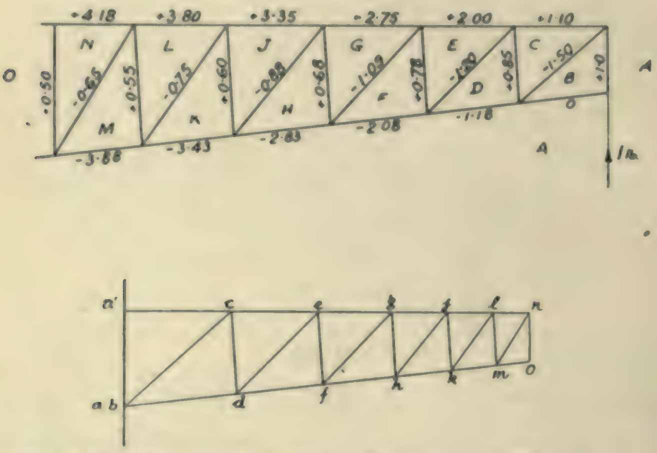


FIG. 56.—TORSION OF REAR PORTION OF FUSELAGE (SIDE PANELS)

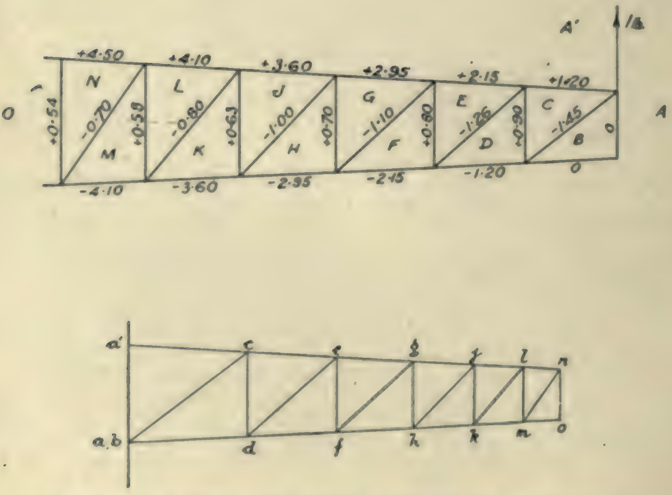


FIG. 57.—TORSION OF REAR PORTION OF FUSELAGE (TOP OR BOTTOM PANELS)

The various factors marked thereon represent the comparative loads in the other members of the structure.

The actual loads in the various members are then obtained by drawing up a table such as Table 19 (inset), which shows the maximum loads in any member for—

(*a*) Side load combined with down load;

(*b*) Side load combined with up load;

and the members must be designed to withstand these maximum loads.

CHAPTER XI

FUSELAGE (*Continued*)

Case 5, 3 Times Down Load

JUST as down load was considered for the wings so its effect on the fuselage must be estimated. It is a case of reversed loading in which the "anti-lift" or "landing" wires take load, the ordinary lift wires being now inoperative. As has been previously stated in Case 3 (Wings), the circumstances under which such a condition of loading may exist are—

(*a*) Possible downward gusts on the wings, or when looping.

(*b*) Reversal of loading in a steep dive.

The difference between these cases being that, for down loading due to downward gusts the centre of pressure is probably at about ·3 of the chord, whereas in the case of a steep dive the centre of pressure is very probably well behind the rear spar, in which latter case there is down load on the front truss and up load in the rear truss.

For the down load case as applied to the fuselage it is usual to assume the C.P. to be at ·3 of the chord, and to take a loading of 3 times load.

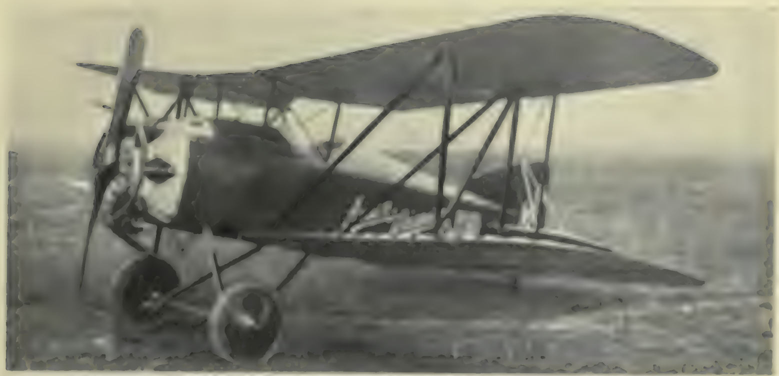
The normal load for the whole machine is therefore 6000 lbs., and this is to be divided between the top and bottom planes according to area and efficiency.

These loads are found to be 1720 lbs. and 1280 lbs. respectively, these being loads for one side of the machine only.

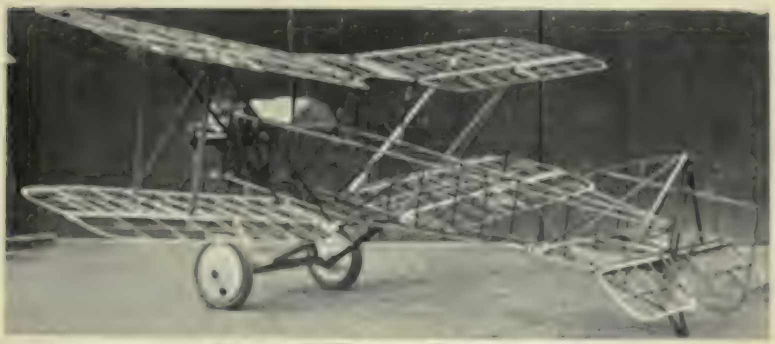
Of the 1280 lbs. on the bottom plane, 1070 lbs. is transferred to the top plane by the anti-lift wires, and is added to the load 1720 lbs. already there, the total load taken on the front and rear body struts being 1070 + 1720 = 2790 lbs., whilst the load on both the bottom plane attachments, front and rear, is 210 lbs.

With C.P. at ·3 of the chord, these loads must be divided proportionately between front and rear spars (*i.e.*, 1950 lbs. and 840 lbs. respectively on top plane, 150 lbs. and 60 lbs. respectively on bottom plane).

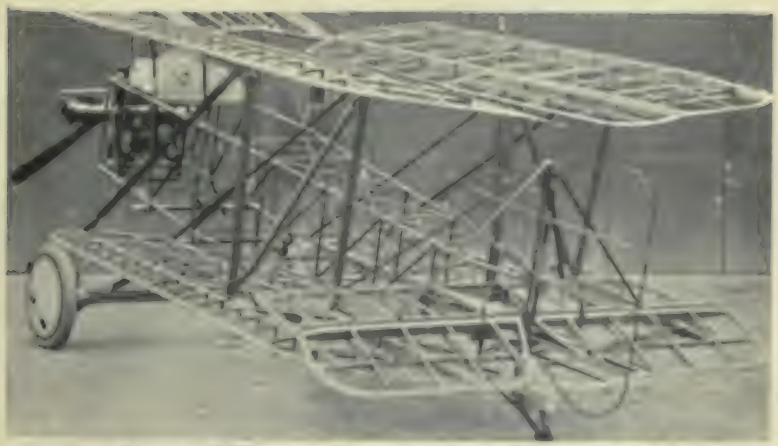
Assuming as before that drag is one-seventh of the lift, then



AUSTIN "WHIPPET" WITH 45-H.P. "ANZANI" ENGINE



AUSTIN "WHIPPET," UNCOVERED



AUSTIN "WHIPPET," UNCOVERED, WINGS FOLDED

the drag load on the top plane is 245 lbs., and on the bottom plane is 185 lbs.

The transference of 1070 lbs. from bottom to top means that it is also moved forward a distance equal to the stagger (*i.e.*, 24 ins.). This introduces a couple of moment 1070 × 24 = 25700 lbs. ins., tending to stall the machine, and since the gap is 56 ins., the two equal horizontal forces of the couple are $\frac{25700}{56} = 460$ lbs., being forward on bottom, and backwards on top plane.

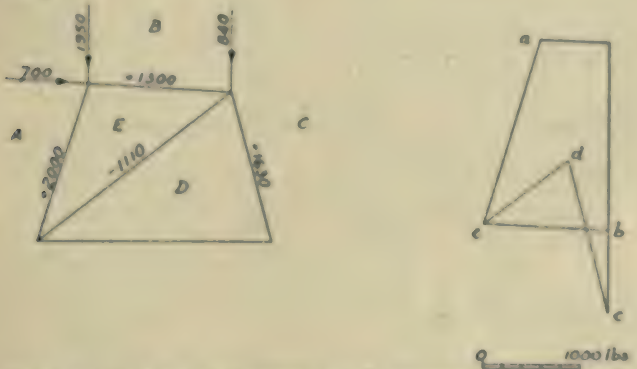


FIG. 58.—DRAG LOADS IN BODY STRUTS AND BRACING DOWN LOAD, CASE 5

Loads in the body struts and side bracing may now be obtained as shown in Fig. 58, these loads being due to drag loads, loads in anti-lift wires, etc.

The loads at the joints of the structure are distributed in exactly the same proportions as in the flying case for the fuselage, but reversed in direction. Generally it is found that an upward tail load is sustained by the tail in this down load case so as to maintain balance, and in order to determine this load, moments about the C.G. should be taken of all the forces operating.

The forces operating on the machine are—

(1) Reversed lift loads.
(2) Tail loads.
(3) Drag forces.
(4) Thrust of propeller.
(5) Reversed dead loads.

Of these, the moments of the thrust and of the reversed dead load are known to be zero.

In Fig. 58 the drag loads and reversed lift loads on the top plane have been combined, and the resulting forces in the body struts and bracing determined.

Taking moments about the C.G. of the whole machine, in order to estimate the tail load—

Moment of load in front body strut	2000	×	24 ins.	=	− 48000
Moment of load in bottom front spar attachment	150	×	9·0 ins.	=	+ 1350
Moment of load in bottom rear spar attachment	60	×	39 ins.	=	+ 2340
Moment of load in side bracing wire	1110	×	16·5 ins.	=	+ 18300
Moment of load in rear body strut	1630	×	23 ins.	=	+ 37500
Moment of bottom drag load	274	×	25·5 ins.	=	+ 7000
Moment of tail load	L	×	181 ins.		

+ and − signs denote stalling and nose-diving moments respectively.

Summing these algebraically—

$$+\ 66490 - 48000 - 181\,L = O$$

$$\therefore\ L = \frac{18500}{181} = 102 \text{ lbs., approximately,}$$

which is an up load, so that the load on the whole tail is 204 lbs.

These reversed lift loads ought now to be increased in the ratio of 3100 to 3000 because of the up tail load.

All that remains now is to find graphically the forces in the various members of the fuselage, as is done in Fig. 59.

CRIPPLING LOADS ON FRAME MEMBERS

From Table 20 the maximum load in every member of the structure may be seen at a glance. For compression members the crippling load as struts must be determined according to the commonly accepted rules. As a first approximation an assumption is made to the effect that the frame joints exercise no fixing effect on the strut.

This is an assumption on the safe side, but how much on the safe side is rather a question; probably 30 to 40%, according to type of fitting. Generally the strut is of such a length compared to its cross-sectional radius of gyration that it cannot be considered as an Euler strut (*i.e.*, $\frac{l}{k} < 120$, where l = length of strut, k = radius of gyration).

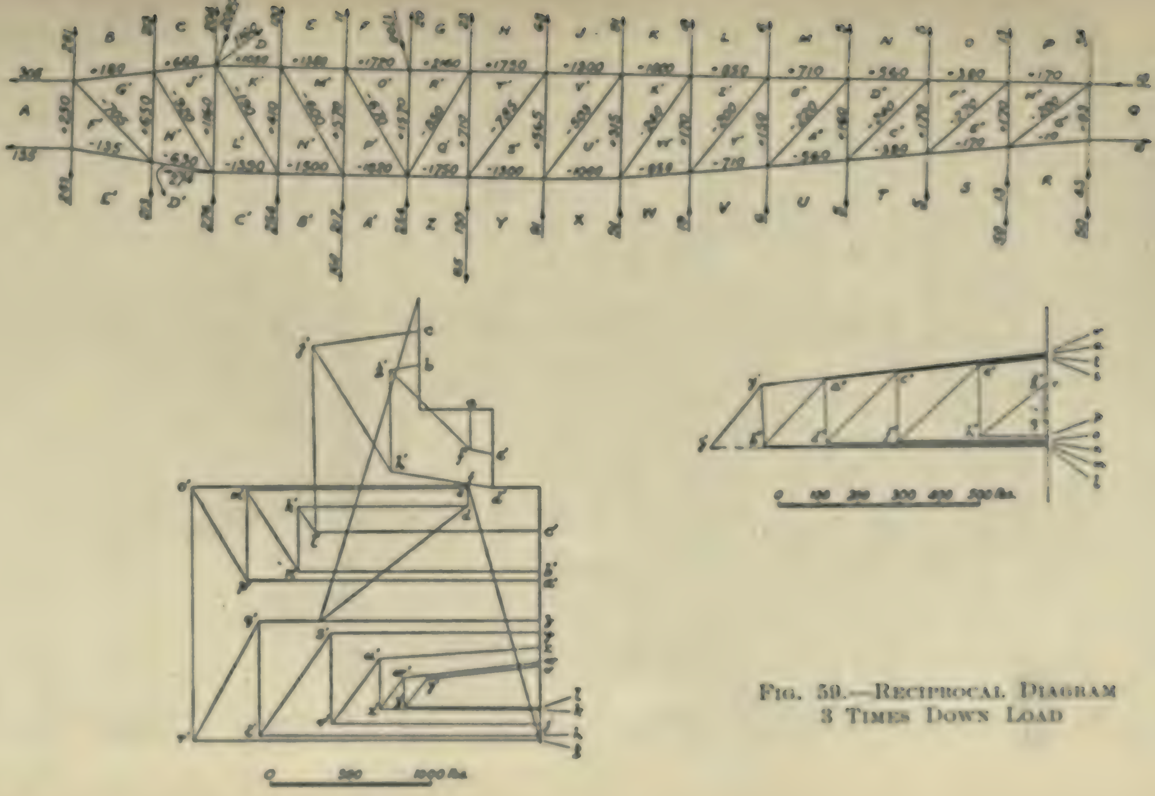


FIG. 59.—RECIPROCAL DIAGRAM
3 TIMES DOWN LOAD

The general method of estimating the crippling load of a strut is therefore: first find the Euler load—

$$P_e = \frac{\pi^2 EI}{l^2}$$

where E = modulus of elasticity in lbs. per sq. ins.;
I = mean moment of inertia of cross section about the major axis; and
l = length between centres in inches.

It should be noted that in the case of a vertical strut the length is assumed to be that between the longeron centres.

Struts are generally tapered in some manner and that is the real reason why the mean value of I is suggested.

Reference to a paper read before the Royal Aeronautical Society by H. A. Webb, M.A., and published in their official organ in April, 1919, will no doubt be helpful and instructive to those interested in the tapering of struts for fuselages, etc.

It must now be estimated what is the value of $\frac{l}{k}$ for the strut, where k is the radius of gyration corresponding to the value of I already used.

Now $Ak^2 = I$; $k^2 = \frac{I}{A}$ $\therefore$ $k = \sqrt{\frac{I}{A}}$

$$\text{and } \frac{l}{k} = l\sqrt{\frac{A}{I}}$$

where A = mean cross-sectional area.

If, as is usual, $\frac{l}{k} < 120$, then the strut is not an Euler strut and the crippling load must therefore be modified as follows—

$$\frac{1}{P} = \frac{1}{P_e} + \frac{1}{P_c}$$

where P is the modified crippling load in lbs., and

$$P_c = f_c \times A,$$

f_c being the compressive stress in lbs. per square inch.

Thus— $$\frac{1}{P} = \frac{P_e + P_c}{P_e P_c} \text{ or, inverting, } P = \frac{P_e P_c}{P_e + P_c}$$

It is possible that a member may fail in tension, in which case the failing load must be—

$$P_t = f_t \times A$$

where f_t is the allowable tensile stress for the material.

In the case of wires, which are always in tension of course, the nearest standard size which will take the load is chosen. Where there are local lateral loads on longerons, these members must be considered in bending with its end load, and if necessary treated as a continuous beam. The bending-moment diagram should be drawn and the deflection diagram obtained by graphical integration in order to estimate the end load bending-moment.

The combined stress must then be found due to bending moment, end load bending-moment, and direct end load.

The crippling loads on steel tubular oval, and circular section struts may easily be found by use of the curves (Figs. 124, 125, 126, 127, 128), which are reprints from Air Ministry tube specifications. These curves are taken from *Recent Researches on Tubular Steel Struts*, and permission to publish them in this volume has been very kindly given by Directorate of Research.

CRIPPLING LOADS OF STRUTS

As an example of the method of estimating the crippling loads of struts take the following—

Maximum actual load on strut $J' L'$ (Table 20, inset), = 3660 lbs.

Let dimensions of strut be 1·45 ins. square in section, the length of the strut being 27 ins.

No taper is assumed in the length of the strut.

The moment of inertia I of the cross section

$$= \frac{1\cdot45 \times 1\cdot45^3}{12} = \cdot368 \text{ ins.}^4$$

then the Euler load is—

$$P_e = \frac{\pi^2\, 1\cdot6 \times 10^6 \times \cdot368}{27^2} = 5820 \text{ lbs.}$$

E being assumed $1\cdot6 \times 10^6$ lbs. per square inch, and $f_c = 5500$ lbs. per square inch.

$$\frac{l}{k} = l\sqrt{\frac{1\cdot45^2}{\cdot368}} = 27\sqrt{\frac{2\cdot1}{\cdot368}} = 64\cdot5 < 120$$

hence the strut is not an Euler strut.

$$f_c \times A = 5500 \times 2\cdot1 = 11550.$$

Then P, the crippling load—

$$= \frac{11550 \times 5820}{11550 + 5820} = 4240 \text{ lbs.}$$

Actual load in strut = 3660 lbs. compression.

∴ Strut is up to strength.

As a further example, take strut $K' N'$ (Table 20).

Length of strut = 27 ins.

Actual maximum load = 1300 lbs.

Strut (parallel) = 1·05 in. square in cross section.

$$\therefore I = \frac{1{\cdot}05 \times 1{\cdot}05^3}{12} = {\cdot}10 \text{ ins.}^4$$

$$P_e = \frac{\pi^2 \times 1{\cdot}6 \times 10^6 \times {\cdot}10}{27^2} = 2170 \text{ lbs.}$$

$$\frac{l}{k} = l\sqrt{\frac{1{\cdot}1}{{\cdot}1}} = 89{\cdot}5$$

∴ Strut is not an Euler strut.

$$f_c \times 1{\cdot}05^2 = 5500 \times 1{\cdot}1 = 6050 \text{ lbs.}$$

$$\text{The crippling load} = \frac{6050 \times 2170}{6050 + 2170} = 1600 \text{ lbs.}$$

∴ Strut is up to strength.

CHAPTER XII

TAIL PLANE

In its capacity as a stabilizing surface, the tail plane may be subjected to either an up load or a down load, and, although the magnitude of the down load is usually in excess of the up load, yet it is advisable that the strength of the tail unit be arranged to suit both circumstances to the same degree.

The centre of pressure on the tail is usually assumed to travel between ·3 and ·5 of the chord. These extreme positions are indicated in Fig. 60.

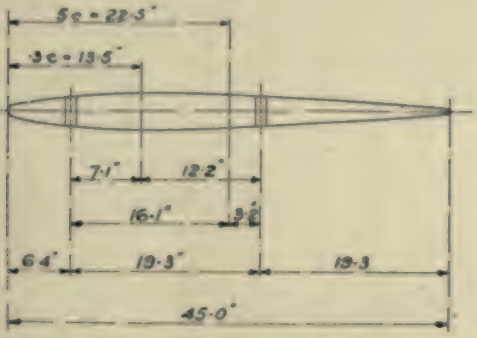


Fig. 60.—Position of Centre of Pressure on Tail Plane

A tail designed to carry its maximum load as given in Fig. 49 with the C.P. in either of these or any other intermediate position, will, in general, be quite safe. The C.P. may move in advance of ·3 of the chord, but this takes place at small angles of incidence when the lift coefficient is small. The tail would not then be called upon to carry its maximum load.

The chord of the tail includes the elevator, being measured from the leading edge of the tail plane to the trailing edge of the elevator.

For tail planes the ratio of lift to drag is usually greater than for main planes (*i.e.*, 4 to 1, as against 7 to 1 for main

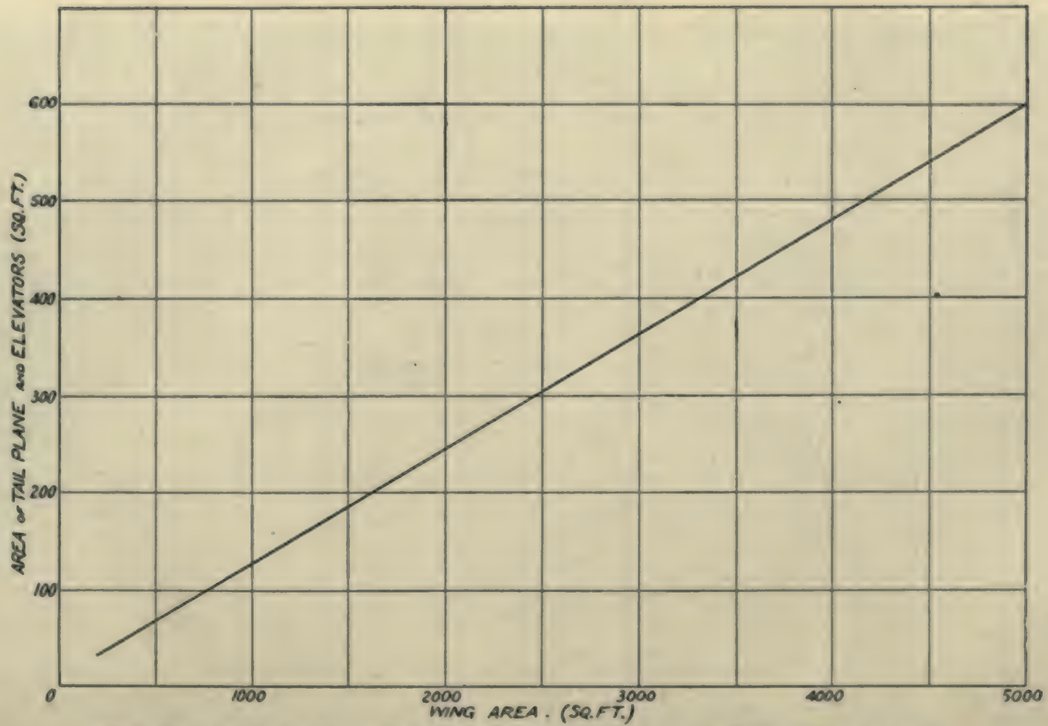


FIG. 61.—AREA OF TAIL PLANE AND, ELEVATORS FOR A GIVEN WING AREA

planes). This is due to the fact that the section of the tail plane is symmetrical and not cambered as an aerofoil section used for main planes.

The maximum down load imposed on the tail is largely dependent upon the rate at which the pilot is able to pull the aeroplane out of a vertical dive, and the maximum up load on the tail is imposed by the pilot in lowering the elevators quickly. These loads are not easy to impose on large and moderately sized machines, but on small aeroplanes with well balanced elevators they are within the power of the pilot to impose. Fig. 61, obtained from actual machines, gives the approximate tail area as 50 sq. ft.

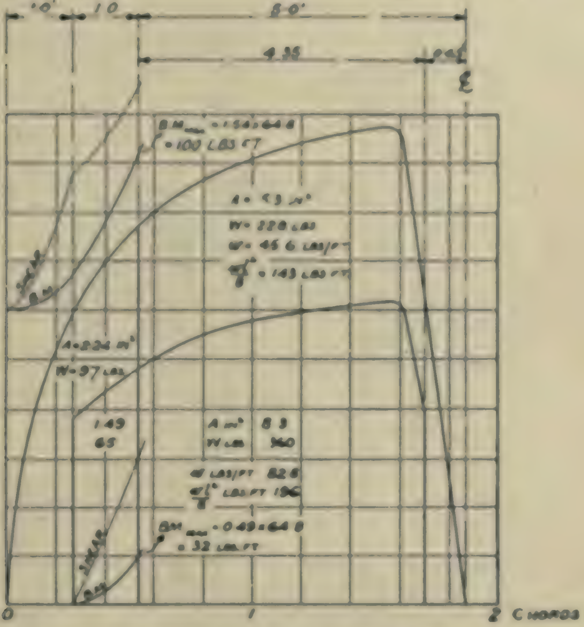


FIG. 62.—LOAD CURVE. FRONT SPAR, C.P. AT ·3 OF THE CHORD

The drag load is, of course, considered simultaneously with the up or down loads. When dealing with the loading on the rear spar, particularly for C.P. at ·5 of the chord, strictly speaking the load on the elevator is transmitted to the rear spar at definite points and not uniformly, these points being the hinge points. In the work which now follows immediately this has been assumed to be uniform.

In a large machine, however, the load must be considered as concentrated at the hinge points instead of being assumed uniform, and if the work already done on the main planes has been thoroughly grasped, there will be no difficulty in applying it here.

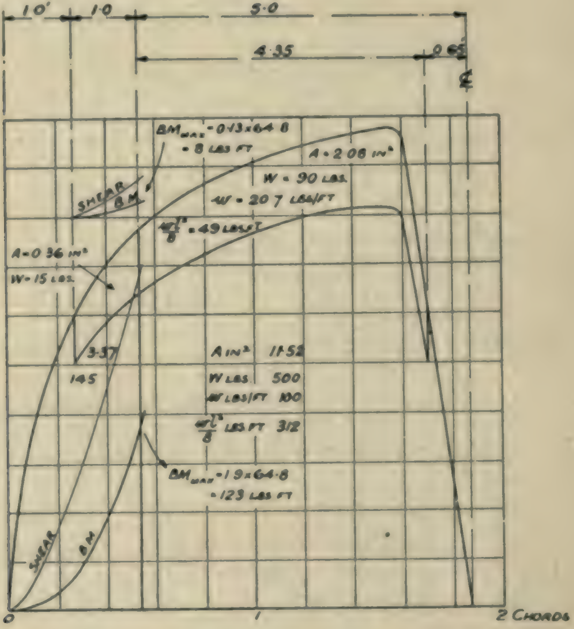


FIG. 63.—LOAD CURVE. REAR SPAR, C.P. AT ·5 OF THE CHORD

Load Curves

The variation of load intensity from the tip of the tail towards the centre is assumed to be exactly similar to that for the main planes, and Figs. 62 and 63 represent the load curves drawn for the tail. The general method of procedure is exactly similar to that already described in connection with the main plane calculations. Using the load curve as drawn in Fig. 62 and taking first the case for centre of pressure at ·3 of the chord, the upper portion represents the load on one-half of the rear spar, and the lower portion represents the load on one-half of the front spar.

The distances between the points of support must be estimated and the diagram divided up accordingly, remembering that, as for the planes, the horizontal scale is 2·5 ins = 1 chord (*i.e.*, 1 in. = 1·5 ft.).

For the purpose of finding the scales of this figure the area under the whole curve must be computed; this has been found to be 17·33 sq. ins. Now the load on one-half the whole tail is $\frac{1500}{2}$ lbs., hence 1 sq. in. of this diagram represents 43·25 lbs.

The mean height of the curve will be found to be

$$\frac{17 \cdot 33}{4 \cdot 66} = 3 \cdot 72 \text{ ins.}$$

The mean load per foot run along the spar

$$= \frac{750 \text{ lbs.}}{7 \text{ ft.}} = 107 \cdot 1 \text{ lbs. per ft.}$$

∴ the load scale is given by

$$1 \text{ in.} = \frac{107 \cdot 1}{3 \cdot 72} = 28 \cdot 8 \text{ lbs. per ft. run,}$$

and the bending-moment scale for the B.M. diagram for the overhang is—

$$1 \text{ in.} = 28 \cdot 8 \times 1 \cdot 5 \times 1 \cdot 5 = 64 \cdot 8 \text{ lbs. ft.}$$

In order to insert on this diagram (Fig. 62) the usual necessary particulars, the separate areas in each bay for portions of the diagram representing both front and rear spars must be estimated, and the corresponding loads calculated.

The overhang bending-moments, and the free bending-moments in the respective bays may be found in the same way as was done for the main planes.

It ought to be noted that this load diagram will do for both up and down load.

In order to ascertain which case of loading will be most severe for the tail plane, it is necessary to roughly investigate the four cases—

(1) C.P. at ·3 of the chord. Up load.
(2) C.P. at ·5 of the chord. Up load.
(3) C.P. at ·3 of the chord. Down load.
(4) C.P. at ·5 of the chord. Down load.

C.P. AT ·3 OF THE CHORD, UP LOAD

Front Spar

The front spar is supported at four points, namely, the two points of attachment of the front external bracing wires, and the two points of attachment of the spar to the sides of the rear fuselage; whereas the rear spar is supported at three points only (*i.e.*, the two points of attachment of the rear external bracing wires, and the point where the rear fuselage converges into the stern post). Thus, on the front spar, there is a short unloaded centre portion passing through the fuselage, but there is no such centre portion on the rear spar.

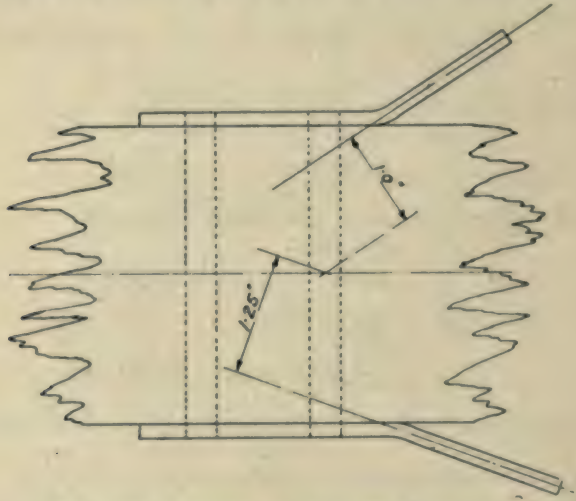


FIG. 64.—OFFSETS OF BRACING WIRES

Free Reactions at Supports

The free reactions at the points of support may be found from the load curve (Fig. 62), but will be modified later by the fixing moments. These free reactions at the inner and outer points of support are 180 lbs. and 245 lbs. respectively.

Offset Bending-Moments

The offset bending-moment may now be estimated, since the approximate load in the front bracing wire is easily calculable, and the offset distance of this wire is shown in Fig. 64. It will be found that the offset bending-moment due to the front bracing wire is 60 lbs. ft. approximately. The overhang bending-moment at the outer support is obtained by a double integration of the load curve, which may be done graphically. Its value will be found to be − 32 lbs. ft.

Preliminary Modified Reactions

These are to be obtained by solving the three moment equations. The bending-moment on the inner side of the support is 60 − 32 = 28 lbs. ft.

Hence, for the front spar—

$$(60 - 32)\,4{\cdot}35 + 2\,M_{2_L} \times 5{\cdot}65 + 1{\cdot}3\,M_{2_L} = -\,2 \times 4{\cdot}35 \times 196$$

$$\text{i.e., } 11{\cdot}3M_{2_L} + 1{\cdot}3M_{2_L} = -\,1700 - 122$$

$$\therefore\ 12{\cdot}6M_{2_L} = -\,1822 \text{ and } M_{2_L} = -\,145 \text{ lbs. ft.}$$

The modified reactions will therefore be—

$$\begin{array}{lll} R_1 = & 65 + 180 - \dfrac{28 + 145}{4{\cdot}35} & = 205 \text{ lbs.} \\ R_2 = & 180 + 40 & = \underline{220} \text{ ,,} \\ & \qquad\text{Total} & = \underline{\underline{425}} \text{ lbs.} \end{array}$$

Rear Spar

In a similar manner, for the rear spar, C.P. at ·3 of chord, the free reactions are 211 lbs. and 228 lbs. at the outer and centre supports respectively. From the upper portion of the load diagram (Fig. 62) the overhang bending-moment will be found to be 100 lbs. ft., and the offset bending-moment due to the rear external bracing wire will be found to be approximately 60 lbs. ft., hence the bending-moment on the inner side of the support will be − 100 + 60 = − 40 lbs. ft.

Hence for the rear spar—

$$(-\,100 + 60)\,5 \times 2 + 2M_{2_L} \times 10 = -\,2 \times 5 \times 143 \times 2$$

$$\therefore\ M_{2_L} = -\,123 \text{ lbs. ft.}$$

The modified reactions will therefore be—

$$\begin{array}{lll} R_1 = & 97 + 114 - \dfrac{123 - 40}{5} & = 194 \text{ lbs.} \\ R_2 = & 228 + 34 & = 262 \text{ lbs.} \end{array}$$

C.P. AT ·5 OF THE CHORD, UP LOAD

Rear Spar

From Fig. 63 the overhang bending-moment is found to be – 123 lbs. ft. The free reactions will be found to be 395 lbs. and 500 lbs. at outer and centre points of support respectively, which enables the offset bending-moment to be determined (*i.e.*, + 100 lbs. ft.). The bending-moment at the inner side of the support will therefore be – 123 + 100 = – 23 lbs. ft. Hence, the three moment equation will be—

$$-23 \times 5 \times 2 + 2\,M_{2_L} \times 10 = -2 \times 5 \times 312 \times 2$$

$$\therefore\ M_{2_L} = -300 \text{ lbs. ft.}$$

The preliminary modified reactions are therefore—

$$R_1 = 145 + 250 - \frac{300 - 23}{5} = 340 \text{ lbs.}$$

$$R_2 = 500 + 110 = 610 \text{ lbs.}$$

Front Spar

The corresponding particulars for the front spar, C.P. at ·5 of the chord are, from Fig. 63—

Overhang bending-moment = – 8 lbs. ft.

The free reactions at outer and inner points of support are 60 lbs. and 45 lbs. respectively, and the offset bending-moment at the outer support will be found to be + 16 lbs. ft. The bending-moment on the inner side of the outer support is therefore 16 – 8 = 8 lbs. ft., and the three moment equation for this spar is—

$$8 \times 4{\cdot}35 + 2\,M_{2_L} \times 5{\cdot}65 + 1{\cdot}3\,M_{2_L} = -2 \times 4{\cdot}35 \times 49$$

$$12{\cdot}6 M_{2_L} = -426 - 35 = -461$$

$$\therefore\ M_{2_L} = -37 \text{ lbs. ft.}$$

The modified reactions at the supports are—

$$R_1 = 15 + 45 - \frac{37 + 8}{4{\cdot}35} = 50 \text{ lbs.}$$

$$R_2 = 45 + 10 = 55 \text{ lbs.}$$

Drag Forces and Force Diagrams, Up Load

Having now roughed out the modified reactions for the cases of up load with centre of pressure in both forward and rear positions, it is possible to construct diagrams showing the loads

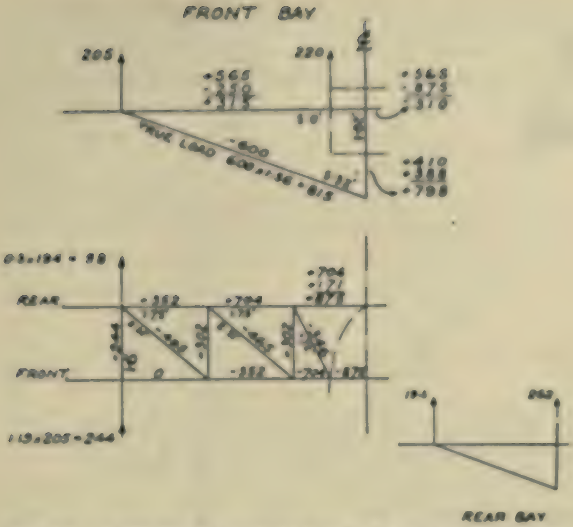


Fig. 65.—Force Diagram C.P. at ·3 of the Chord. Up Load

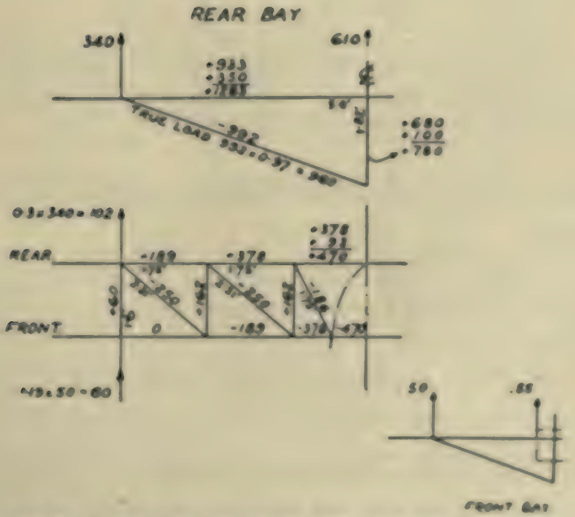


Fig. 66.—Force Diagram, C.P. at ·5 of the Chord. Up Load

in the bracing (internal and external), and the tail plane spars. This has been done in Figs. 65 and 66, where it will be noticed that the drag forces have been inserted along with the up load reactions. These drag forces are determined graphically

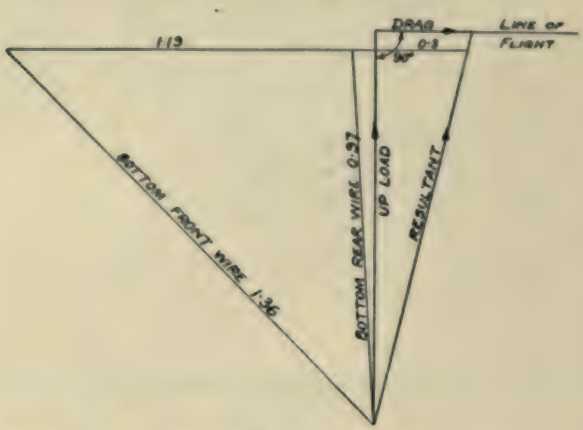


FIG. 67.—DRAG COEFFICIENTS DIAGRAM. UP LOAD

by means of the drag coefficient diagram (Fig. 67), which corresponds to the ones constructed for estimating the wing load drags, and needs no further description here.

C.P. AT ·3 OF THE CHORD, DOWN LOAD

Front Spar

Similar particulars to those obtained for the up load case are required for this down load case, and are derived from Fig. 62, namely—

Overhang bending-moment = − 32 lbs. ft.

Free reactions at outer and inner supports are 245 lbs. and 180 lbs. respectively, which enables the offset bending-moment, due to the upper external bracing wires being offset 1 in., as shown in Fig. 64, to be estimated.

This will be found to be + 38 lbs. ft. at the outer support. The bending-moment on the inside of this support is therefore 38 − 32 = 6 lbs. ft.

The three moment equation to obtain M_{2_L} is therefore—

$$6 \times 4{\cdot}35 + 12{\cdot}6\, M_{2_L} = -2 \times 4{\cdot}35 \times 196$$

$$12{\cdot}6\, M_{2_L} = -1700 - 26$$

hence $$M_{2_L} = -137 \text{ lbs. ft.}$$

The modified reactions at the outer and inner points of support become—

$$R_1 = 65 + 180 - \frac{137 + 8}{4{\cdot}35} = 212 \text{ lbs.}$$

$$R_2 = 180 + 33 = \underline{213 \text{ lbs.}}$$

$$\text{Total} = \underline{\underline{425 \text{ lbs.}}}$$

Rear Spar

From Fig. 62 the overhang bending-moment at the outer support is − 100 lbs. ft., and the free reactions for the outer and central supports are 211 lbs. and 228 lbs. respectively.

The offset bending-moment at the outer support is therefore + 36 lbs. ft., so that the bending-moment on the inner side of the outer support is − 100 + 36 = − 64 lbs. ft.

The equation of three moments for the rear spar for this case is—

$$-64 \times 5 \times 2 + 2\, M_{2_L} \times 10 = -2 \times 5 \times 143 \times 2$$

$$\therefore M_{2_L} = -111 \text{ lbs. ft.}$$

The modified reactions are therefore—

$$R_1 = 97 + 114 - \frac{111 - 64}{5} = 202 \text{ lbs.}$$

$$R_2 = 228 + 18 = 246 \text{ lbs.}$$

C.P. AT ·5 OF THE CHORD, DOWN LOAD

Rear Spar

From Fig. 63 the overhang bending-moment is − 123 lbs. ft., and the free reactions are 395 lbs. and 500 lbs. at the outer and centre points of support respectively.

The offset bending-moment + 63 lbs. ft., so that the bending-moment on the inner side of the outer support is—

$$-123 + 63 = -60 \text{ lbs. ft.,}$$

hence the three moment equation will be—

$$-60 \times 5 \times 2 + 2\, M_{2_L} \times 10 = -2 \times 5 \times 312 \times 2$$

$$\therefore M_{2_L} = \underline{-286 \text{ lbs. ft.}}$$

Modified reactions at outer and centre points of support are therefore—

$$R_1 = 145 + 250 - \frac{286 - 6}{5} = 350 \text{ lbs.}$$

$$R_2 = 500 + \ \ 90 \qquad\qquad = 590 \text{ lbs.}$$

Front Spar

From Fig. 63, again, the overhang bending-moment is − 8 lbs. ft., and the free reactions are 60 lbs. and 45 lbs. at the outer and inner points of support respectively.

The offset bending-moment is + 9 lbs. ft., hence the bending-moment on the inner side of the outer support is—

$-8 + 9 = +1$ lbs. ft.

Equation of three moments—

$$1 \times 4{\cdot}35 + 12{\cdot}6\, M_{2_L} = -426$$

$$12{\cdot}6\, M_{2_L} = -430$$

$$M_{2_L} = -\ 34 \text{ lbs. ft.}$$

The modified reactions are—

$$R_1 = 15 + 45 - \frac{34 + 1}{4{\cdot}35} = 52 \text{ lbs.}$$

$$R_2 = 45 + \ \ 8 \qquad\qquad = 53 \text{ lbs.}$$

Drag Forces and Force Diagrams, Down Load

Force diagrams showing the loads in spars and bracing wires may now be constructed as in Figs. 68 and 69.

The drag forces are determined from the drag coefficient diagram (Fig. 70), and are inserted in Figs. 68 and 69 in order to estimate the loads in the various members of the structure.

An examination of the Figs. 65, 66, 68, and 69 will reveal the fact that the up load case is the one which induces larger stresses in the structure generally, and so it is proposed to neglect the down load case at this point and devote all the attention to that of the up load.

This conclusion might have been arrived at earlier by an experienced designer, since it would have been observed that the angles of the external bracing wires on the under side of the tail plane are smaller than those of the bracing wires on the upper side, thus producing heavy end load in the spars.

However, the force diagrams (Figs. 68 and 69) are necessary

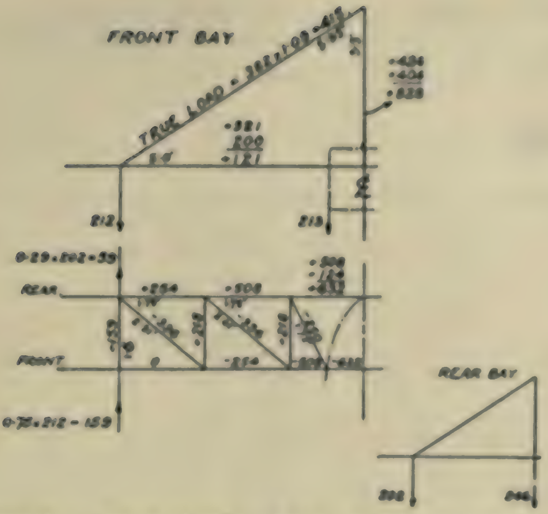


Fig. 68.—Force Diagram C.P. at ·3 of the Chord.
Down Load

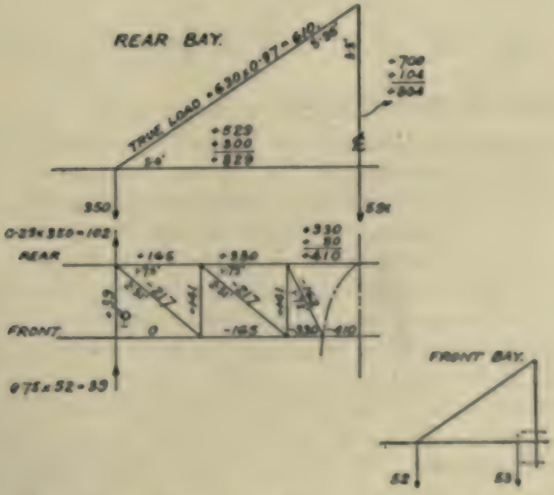


Fig. 69.—Force Diagram C.P. at ·5 of the Chord.
Down Load

in order to obtain the loads in the upper external wires, but the spar stresses are only found for the up load case which covers the other case in that respect.

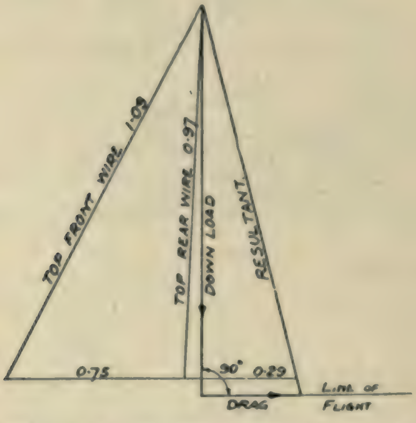


Fig. 70.—Drag Coefficient Diagram. Down Load

C.P. AT ·3 OF THE CHORD, UP LOAD

Final Calculations, Front Spar

Having decided that the up load case is the one producing the maximum stresses in the spars, then the calculations essential to the estimation of the stresses ought now to be made. The procedure is exactly similar to that for the wing cases where the Webb-Thorne method was applied.

Table 22 is a tabulation of the quantities leading up to the three moment equations for the front spar using Webb-Thorne method, where it will be seen that the same value of *E* has been assumed for the material.

The value of *I* is obtained by the graphical method previously recommended, and it will be noted that the tail plane section adopted is such that the depth of both front and rear spars is the same.

At this point a trial guess would be made of the spar sections, and this may be done very closely after some experience has been gained. Fig. 71 shows the section decided upon in this case, where the values of area, moment of inertia, and modulus of the section are also given.

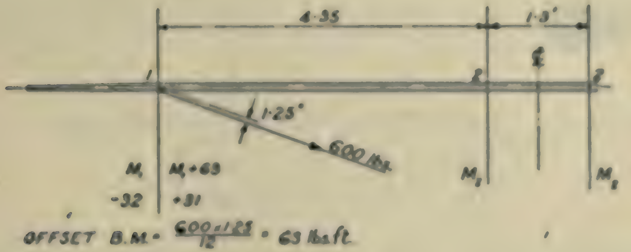

4·35
1·3'
1
2
2
℄
1·25"
600 lbs.
M_1
$M_1 + 63$
−32
+31
M_2
M_2
OFFSET B.M. = $\frac{600 \times 1{\cdot}25}{12}$ = 63 lbs. ft.

TABLE 22

Span (feet) . .	4·35	1·3
Span (ins.) . .	52·2	15·6
I in.^{4} . . .	1·76	1·76
$P_e = \frac{\pi^2 EI}{l^2}$ . .	10,250	115,000
End load . .	+ 315	− 310
$Pe - P$ . . .	9935	115,310
w lbs. per ft. .	82·8	0
$\frac{wl^2}{4}$	392	0
$\frac{P}{P_e}$	·0307	·00269
$1 + 0{\cdot}2\frac{P}{P_e}$ . .	1·00	1·00
$1 - 0{\cdot}38\frac{P}{P_e}$ . .	1·00	1·00
$1 - \frac{P}{70 P_e}$ . .	1·00	1·00
$\frac{1}{(P_e - P)l}$ . .	$2{\cdot}31 \times 10^{-5}$	$\cdot 668 \times 10^{-5}$
$M_R \quad M_L$ . .	+ 31 − 147	− 147 − 147
$\frac{1}{2}(M_R + M_L)$ . .	− 58	− 147
$1 + \cdot 26\frac{P}{P_e}$ . .	1·00	1·00
$1{\cdot}02\left(\frac{wl^2}{8}\right)$ . .	200	0
$(M_L - M_R)$ . .	− 178	0
$\frac{(M_L - M_R)^2}{2\,wl^2}$ . .	10·1	0

On the completion of the tabulation of Table 22, the final modified reactions are determined by solving the three moment equations as follows, from (1)* in Appendix.

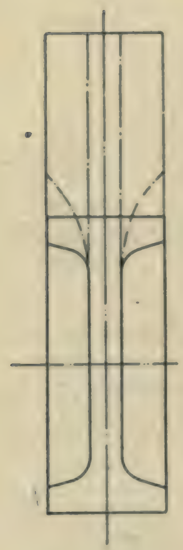
$$31 \times 1 \times 2{\cdot}3 \times 10^{-5} + 2\,M_{2_L} \times 1 \times 2{\cdot}3 \times 10^{-5} + 392 \times 1 \times 2{\cdot}31 \times 10^{-5} + M_{2_L} \times 1 \times {\cdot}668 \times 10^{-5} + 2\,M_{2_L} \times 1 \times {\cdot}66 \times 10^{-5} = O$$

then

$$4{\cdot}62\,M_{2_L} + {\cdot}668\,M_{2_L} + 1{\cdot}336\,M_{2_L} = -905 - 72$$

$$6{\cdot}624\,M_{2_L} = -977$$

$$\therefore\ M_{2_L} = -147 \text{ lbs. ft.}$$



SECTION	AREA OF SECTION	AREA OF CUBE FIG.	MI IN^4	Z IN^3
SOLID	3·60	4·05	2·70	1·80
SOLID AND 5/16" HOLE	2·66	3·00	2·00	1·33
HOLLOW	1·60	2·65	1·76	1·17

FIG. 71
PROBABLE SPAR SECTION

The final corrected reactions are therefore—

$$R_1 = 65 + 180 - \frac{28 + 147}{4{\cdot}35} = 205 \text{ lbs.}$$

$$R_2 = 180 + 40 = 220 \text{ lbs.}$$

which by comparison with those preliminarily estimated will be found to be identical. Hence the diagram (Fig. 65) remains unaltered.

---

* Figures refer to formulæ numbers in Webb-Thorne method in Appendix.

Bending-Moment Diagram

The bending-moment at various points along the spar inward from the outer support should now be computed in the same way as was done for the wing spars. Thus, the bending-moment at the mid point of the bay is given by equation (2) in the Appendix.

$$M_{mid} = \frac{10250}{9935}(-58 \times 1{\cdot}0 + 200)$$

$$= \frac{10250}{9935} \times 142 = 147 \text{ lbs. ft.}$$

and the maximum bending-moment is given by (3) Appendix —

$$M_{max} = 147 + 10 = 157 \text{ lbs. ft.}$$

the distance a is given by (4) Appendix

$$2{\cdot}175 + \frac{-178}{82{\cdot}8 \times 4{\cdot}35} = 2{\cdot}175 - {\cdot}495 = 1{\cdot}68 \text{ ft.}$$

$$c_2 = 2{\cdot}175 - {\cdot}9 \times {\cdot}495 + 1{\cdot}2\sqrt{\frac{157}{82{\cdot}8}} \text{ from (5b) Appendix}$$

$$= 2{\cdot}175 - {\cdot}445 + 1{\cdot}65 = 3{\cdot}38 \text{ ft.}$$

Realizing that the bending-moment is constant along the centre bay, and that the overhang bending-moment may be copied from the load curve (Fig. 62), it is now possible to draw the bending-moment diagram for the spar by joining up the points already found by calculation (*see* Fig. 72).

TABLE 23

Section	A in.^{2}	Z in.^{3}	B.M. lbs. ft.	End load lbs.	B.M. Stress, lbs./in.^{2}	E.L. Stress. lbs/in.^{2}	Total Stress. lbs./in.^{2}	F/S.
a	1·6	1·17	18	—	185	—	185	O.K.
1*a*	2·66	1·33	32	—	290	—	290	„
1*b*	2·66	1·33	31	+ 565	280	+ 210	490	„
b	1·6	1·17	65	+ 565	670	+ 350	1020	„
c	1·6	1·17	157	+ 565	1610	+ 350	1960	„
d	1·6	1·17	100	− 139	1025	− 85	940	„
2*d*	2·66	1·33	147	− 139	1325	− 50	1275	„
2*e*	2·66	1·33	147	− 310	1325	− 115	1210	„
e	1·6	1·17	147	− 310	1505	− 195	1310	„

Table of Stresses

On the diagram Fig. 72 the limits of spindling of the spar are indicated by *a*, *b*, *d*, etc., and a table of the stresses has been made (Table 23), from which it will be seen that the stress is not excessive at any point.

Shear Due to Bending

From Fig. 72 it is seen that the shear force is a maximum at point *d*, its value, *F*, obtained from slope of bending-moment diagram, being 180 lbs.

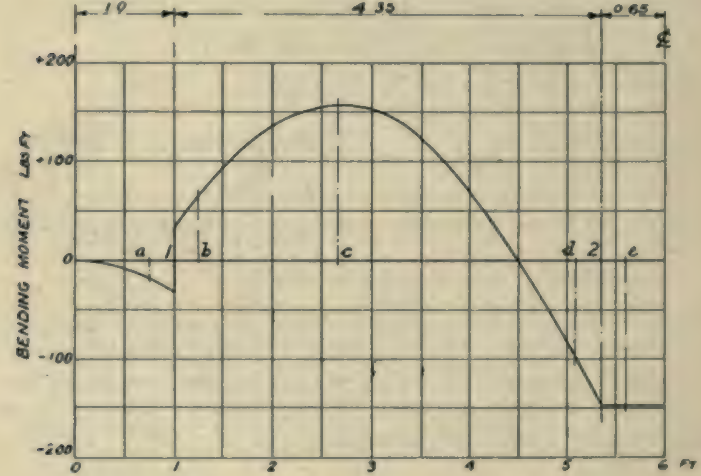


FIG. 72.—BENDING-MOMENT DIAGRAM. FRONT SPAR C.P. AT ·3 OF THE CHORD. UP LOAD

The shear stress—

$$q = \frac{180}{1{\cdot}76}\left\{\left(\frac{1{\cdot}2}{{\cdot}312} \times \frac{3^2 - 2{\cdot}375^2}{8} + \frac{2{\cdot}375^2}{8}\right)\right\}$$

$$= \frac{180}{1{\cdot}76}\left\{\frac{1{\cdot}2}{{\cdot}312} \times \frac{3{\cdot}37}{8} + {\cdot}704\right\}$$

$$= \frac{180}{1{\cdot}76} \times 2{\cdot}324$$

$= 240$ lbs. per in.^{2} which is quite allowable.

C.P. AT ·5 OF THE CHORD, UP LOAD

Final Calculations, Rear Spar

The procedure of calculating the stresses in the rear spar does not differ in any way from that obtaining in the case of the front spar which has just been completed.

Table 24 must be compiled, followed by the solution of the equations as follows, which give, (1) the fixing moments, and (2) the bending-moment at various points along the spar.

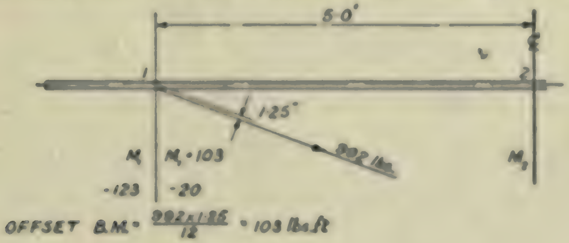


TABLE 24

Span (ft.) . . .	5·0
Span (ins.) . . .	60
I in.^{4}	1·76
$P_e = \dfrac{\pi^2 EI}{l^2}$ . . .	7780
End load (comp.) .	1283
$P_e - P$	6497
w lbs. per ft. . .	100
$\dfrac{wl^2}{4}$	624
$\dfrac{P}{P_e}$	·165
$1 + 0{\cdot}2\dfrac{P}{P_e}$ . . .	1·033
$1 - 0{\cdot}38\dfrac{P}{P_e}$ . . .	0·9373
$1 - \dfrac{P}{70P_e}$	0·9976
$\dfrac{1}{(P_e - P)l}$	$3{\cdot}08 \times 10^{-5}$
$M_R \quad M_L$ . . .	$-20 \quad -305$
$\frac{1}{2}(M_R + M_L)$ . . .	$-162{\cdot}5$
$1 + {\cdot}26\dfrac{P}{P_e}$ . . .	1·0429
$1{\cdot}02\left(\dfrac{wl^2}{8}\right)$ . . .	318
$(M_L - M_R)$ . . .	-285
$\dfrac{(M_L - M_R)^2}{2\,wl^2}$ . . .	16·25

The equation for the fixing moment at the centre point of support becomes

$$-20 \times 1{\cdot}035 \times 3{\cdot}08 \times 10^{-5} \times 2 + 2\,M_{2_L} \times {\cdot}9373 \times 3{\cdot}08$$
$$\times\, 10^{-5} \times 2 + 624 \times {\cdot}9976 \times 3{\cdot}08 \times 10^{-5} \times 2 = O$$
$$1{\cdot}9746\,M_{2_L} = -623 + 21 = -602$$
$$\therefore M_{2_L} = -305 \text{ lbs. ft., from (1) in Appendix.}$$

The final corrected reactions are therefore

$$R_1 = 145 + 250 - \frac{305 - 23}{5} = 340 \text{ lbs.}$$
$$R_2 = 500 + 110 = 610 \text{ lbs.}$$

These reactions are practically the same as those already preliminarily estimated for this case, and so the diagram (Fig. 66) showing the loads in external and internal bracing and end loads in spars remains unaltered.

Bending-Moment Diagram

As is done for the front spar the bending-moments at several points may now be found.

Utilizing the several formulae in the Appendix, as was done for the front spar, then—
the bending-moment at the mid point of the inner bay is

$$M_{mid} = \frac{7780}{6497}(-162{\cdot}5 \times 1{\cdot}0429 + 318)$$
$$= \frac{7780 \times 148}{6497} = 177 \text{ lbs. ft.}$$

and the maximum bending-moment in that bay is given by—

$$M_{max} = 177 + 16 = 193 \text{ lbs. ft.}$$

$$\text{The distance } a = 2{\cdot}5 - \frac{285}{100 \times 5}$$
$$= 2{\cdot}5 - {\cdot}57 = 1{\cdot}93 \text{ ft.}$$

$$\text{the distance } c_2 = 2{\cdot}5 - {\cdot}9 \times {\cdot}57 + 1{\cdot}2\sqrt{\frac{193}{100}}$$
$$= 2{\cdot}5 - {\cdot}513 + 1{\cdot}67 = \underline{3{\cdot}657 \text{ ft.}}$$

A diagram showing the bending-moment at any point along the spar may now be drawn.

Fig. 73 shows the diagram and indicates the points *a*, *b*, *d*, where the spindling terminates.

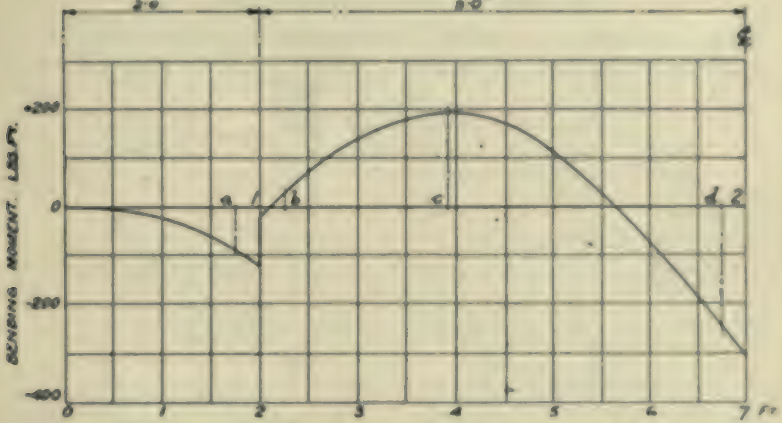


FIG. 73.—BENDING-MOMENT DIAGRAM, REAR SPAR C.P. AT ·5 OF THE CHORD. UP LOAD

Table of Stresses

These are tabulated in Table 25, from which it will be observed that the stress is not excessive anywhere.

TABLE 25

Section.	*A* in.^{2}	*Z* in.^{3}	B.M. lbs. ft.	End Load lbs.	B.M. Stress. lbs./in.^{2}	E.L. Stress. lbs./in.^{2}	Total Stress. lbs./in.^{2}	F/S.
a	1·6	1·17	90	—	920	—	920	O.K.
1*a*	2·66	1·33	123	—	1110	—	1110	,,
1*b*	2·66	1·33	20	1122	180	420	600	,,
b	1·6	1·17	30	1122	310	700	1010	,,
c	1·6	1·17	193	1311	1980	820	2800	,,
d	1·6	1·17	204	1403	2460	880	3340	,,
2*d*	2·66	1·33	305	1403	2750	530	3280	,,

Shear Due to Bending

From the bending-moment diagram (Fig. 73), this is a maximum at *d*.

The shear force F is found to be 245 lbs., and therefore

$$q = \frac{245}{1 \cdot 76} \times 2 \cdot 324 = 325 \text{ lbs. per in.}^2 \text{ which is allowable.}$$

NOTE ON TAIL SECTIONS

Fig. 74 shows two shapes of section which are in present use. If the resistance of the tail can possibly be minimized by the use of a particular section, then the designer will use his discretion as to its adoption. The section, however, must have sufficient depth to admit of a sufficiently strong spar being inserted. Other shapes in use are aerofoil sections inverted for seaplanes and flying boats where down lift is desired, to balance the propeller thrust moment about centre of gravity.

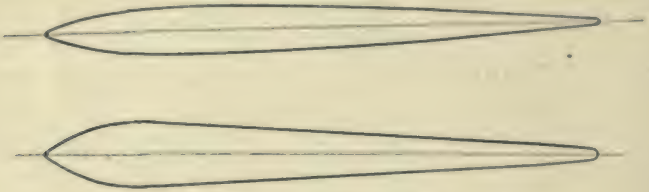
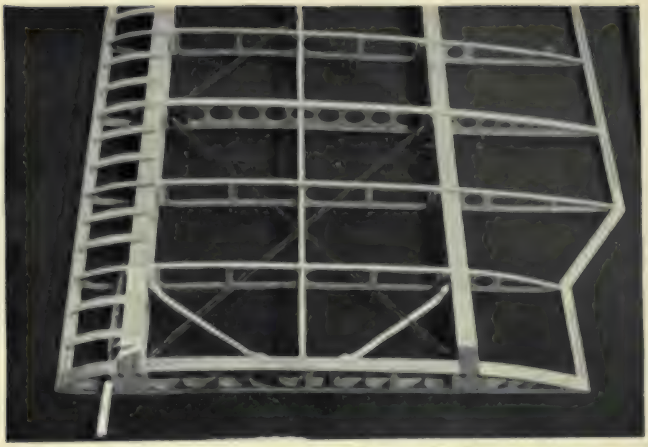
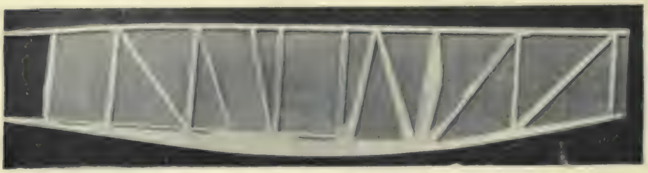


FIG. 74.—SHAPES OF TAIL SECTIONS

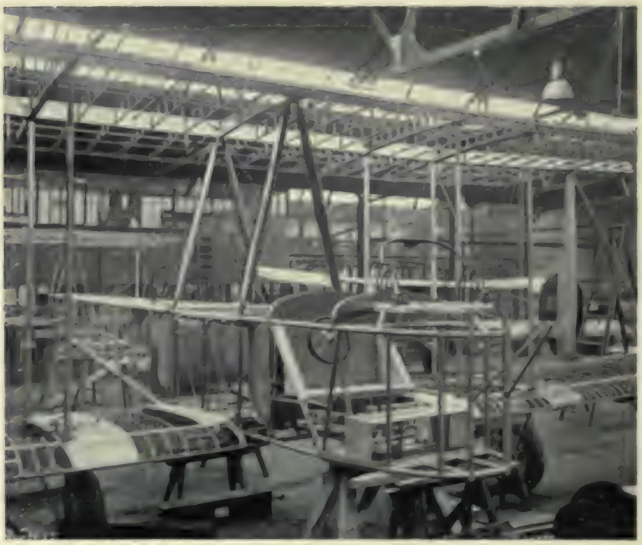
PLATE 6



WING CONSTRUCTION FOR AIRCO 9 AEROPLANE



SIDE OF FUSELAGE FOR AIRCO 9 AEROPLANE



VICKERS' "VIMY" BOMBER, SHOWING CONSTRUCTION

CHAPTER XIII

LANDING GEAR

The whole shock of landing may either be taken on the landing gear alone, as is usual, or it may be proportionately divided between the wheels and the rear skid. Experience has shown, however, that if the landing gear is designed so as to withstand the load due to landing on the wheels alone, then it will be quite satisfactory for the second case. When landing on the wheels, the attitude of the aeroplane will be such that the load reaction at the point of contact of the wheels and the ground is in such a direction that it passes through the centre of gravity of the whole machine, but, as a general rule, it is quite satisfactory if the direction of the load reaction is perpendicular to the wing chord.

The usual factor is that of 5 (*i.e.*, the landing gear must withstand 5 times the static weight of the fully loaded machine), so that each side of the landing gear must withstand 2½ times the weight of the machine. Of the several types of landing gear, the one known as the V type is the most popular, and this is the type to be considered in these calculations.

In this type, as a general rule, the front struts only are braced. Using a factor of 5, the load at each wheel when the aeroplane is landing on its wheels alone is 5830 lbs., the weight of the whole machine being 2330 lbs.

Two or three projected views of the landing gear ought now to be drawn, in outline only, as is done in Fig. 75. Treating the figure as a simple space structure and applying the method of estimating the loads known as the "Gauche" method of solution, in which the force polygon consists of two views, the one projected from the other, and whose outlines are respectively parallel to the various members in the two projected views in the position diagram.

The two views in the position diagram which have been chosen are the side and front elevations, as will be seen in Fig. 75.

The drawing of the force polygons will, no doubt, be easily followed from this same figure. Since neither of the force polygons necessarily shows true lengths, it follows that, in order to read off the actual forces in the various members, a composition diagram showing true lengths must be drawn.

Actually the force polygons in Fig. 75 represent two distinct cases of landing—

(*a*) Landing on both wheels simultaneously, in which case there is no load imposed in the cross bracing, and the force polygon is represented by 1, 2, 3, 4, but there is some tension in the axle or cross tubes.

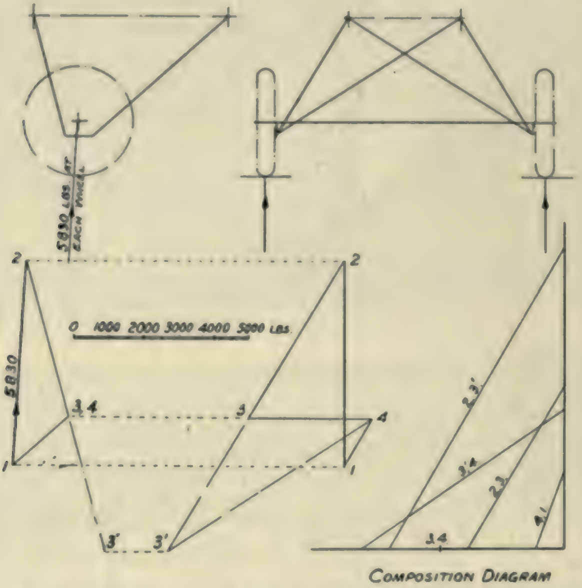


FIG. 75.—LANDING GEAR. DIAGRAM OF FORCES

(*b*) Landing on one wheel when there is load in the cross bracing and none in the cross tubes, the force polygon being represented by 1, 2, 3′, 4.

From the composition diagram, a table of loads has been drawn up for the two cases, as follows, tensile loads being shown negative—

MEMBER.	Landing on both wheels at 5 times load.	Landing on one wheel at 2½ times load.
Front strut, 2, 3 . . .	5330 lbs.	9820 lbs.
Rear strut, 4, 1 . . .	2250 lbs.	2250 lbs.
Axle, 3, 4	− 3500 lbs.	zero
Bracing wire, 3′, 4. . .	zero	− 6940 lbs.

LANDING GEAR STRUTS

These may be either of streamline spruce or streamline metal tube, or again a common form is the round metal tube with wooden 3 to 1 fineness ratio fairing attached, when, according to the size of this fairing, the member may take much more compressive load.

Assuming that wooden struts are to be utilized, then the following method may be used.

Front Strut

Maximum load in strut = 9820 lbs. Approximate length = 34·5 in. Pin jointed ends assumed. Strut assumed to be made of spruce and 3 to 1 parallel streamline section.

$$\therefore I = \cdot 132\, t^4 \text{ where } t = \text{maximum thickness.}$$

$$9820 = \frac{\pi^2\, 1{\cdot}6 \times 10^6 \times \cdot 132\, t^4}{34{\cdot}5^2}$$

$$\therefore t^4 = \frac{9820 \times 34{\cdot}5^2}{\pi^2 \times 1{\cdot}6 \times 10^6 \times \cdot 132} = 5{\cdot}61 \text{ in.}^4$$

$$\therefore t = 1{\cdot}54 \text{ ins.}$$

The value of $\frac{l}{k}$ for this strut will be found to be about 100, so that, strictly speaking, the strut will fail at a load less than 9820 lbs., as explained on page 142.

In view of the fact that the strut is being designed for a severe case, the value of t just found would be sufficiently accurate to meet the conditions of the case.

The dimensions of the strut section will be, therefore, 1·54 ins. × 4·62 ins.

Rear Strut

Maximum load in strut, 2250 lbs. Approximate length = 48 ins. Other conditions as for front strut.

$$\therefore I = \cdot 132\, t^4$$

$$2250 = \frac{\pi^2\, 1{\cdot}6 \times 10^6 \times \cdot 132\, t^4}{48^2}$$

$$\therefore t^4 = \frac{2250 \times 48^2}{\pi^2\, 1{\cdot}6 \times 10^6 \times \cdot 132} = 2{\cdot}47$$

$$\therefore t = 1{\cdot}25 \text{ ins.}$$

The value of $\frac{l}{k}$ is 155, so that this is an Euler strut, and therefore—

Its section will be 1·25 ins. × 3·75 ins.

If circular tubes be used, then the same method applies, except that I is now $= \frac{\pi(D^4 - d^4)}{64}$ where D and d are the outer and inner diameters.

If it is decided to use parallel streamline tubular struts, then it should be remembered that for a 3 to 1 ratio it may be assumed that such a strut has the same strength as a circular tubular strut, whose diameter is 1·5 times the minor axis of the streamline section, other things being equal.

Landing Gear Axle

The bending-moment on the axle is constant between the points of attachment to the landing gear struts and equal to 5830 × 5 ins. = 29150 lbs. ins.; 5 ins. being the mean overhang of the wheel.

Using axle tubing $1\frac{3}{4}$ O.D. × 11 S.W.G., and Air Board or British Standards specification $T2$, the ultimate bending and tensile stresses of which are about 80 and 85 tons per square inch respectively.

The modulus of the tube section will be found to be ·2283, hence the bending stress

$$= \frac{29150}{\cdot 2283} = 128{,}000 \text{ lbs. per sq. ins.}$$

which is quite allowable.

Cross Bracing Wire

A $\frac{5}{16}$ in. B.S.F. streamline wire (5600 lbs.) would probably be found to give no trouble. The condition that the wire takes 6940 lbs. is rather severe, and this load would not be reached when merely taxi-ing sharply round or running over an average aerodrome.

REAR SKID

The skid considered here is of the all-steel type, where helical springs are used for absorbing the shock at landing.

The problem of estimating the loads in the various members is attacked in practically the same way as if rubber shock absorbers were used.

As in the case of the landing gear, the skid should stand up to about 5 times its static load.

When the machine is on the ground the load taken by the skid is easily found when the position of the C.G. is known. In the present case it is 240 lbs., which at 5 times load becomes 1200 lbs., and is assumed to act normal to ground. Fig. 76 shows the skid in its fully extended position. It is seen that the vertical skid post is hinged to the bottom fin tube, a lever being fixed to its upper end. The lever is connected by a wire to the rudder bar, thus making the skid steerable. It is not advisable to use the rudder control cables for this purpose, as the cable might get strained to such an extent whilst being used for steering, preparatory to taking off, that when in the air the rudder control loads might be sufficient to break the cable, rendering the rudder useless.

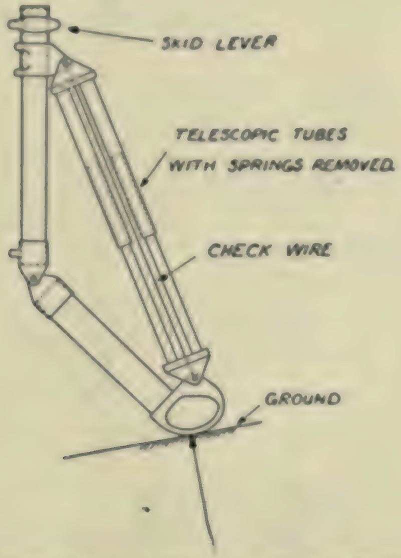


Fig. 76.—General Arrangement of Rear Skid. Fully Extended Position

The skid shoe is connected to the skid arm, which is pinned to the lower end of the skid post. Two springs are used. These are placed between the top of the skid shoe and the top of the skid post, at which places sockets are arranged as indicated.

The springs are held side by side in the following manner. Two tubes about 8 ins. long, which will just fit inside the springs, are brazed in position on the top socket.

Similarly, tubes are brazed to the bottom socket, but of a smaller size, so that they will telescope into the two fitted to the top socket. Thus, when the springs are in position, the tubes act as guides and at the same time allow the spring to compress.

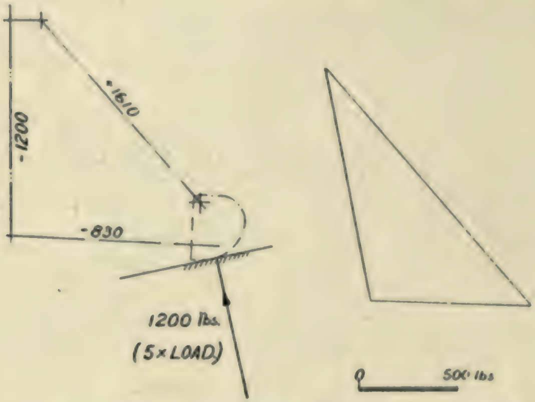


Fig. 77.—Force Diagram for Skid. Closed Position

In the normal position the springs are given a small initial tension by means of a short piece of cable connecting the top and bottom sockets.

In order to obtain the loads in the members, the normal position of the skid should not be taken, for as soon as load is applied, the springs compress. Thus, in the case of elastic absorbers the position of the skid for various loads must be found by trial and error, as the extension of the elastic is not directly proportional to the load. In the case of the spring, however, the deflection can be obtained at once for a given spring and known loads.

Set out the skid as in Fig. 77, assuming a 4 ins. deflection of the springs. The load taken by them in this position is 1610 lbs., or 805 lbs. each. Thus, a spring is required which will carry a load of about 800 lbs. at 4 ins. deflection.

Suppose the springs were of 4 gauge steel (·232 ins. diameter wire), mean radius of coils ·45 ins., 35 working coils.

The rate of the spring or the load required to compress it 1 in. is given by

$$\frac{N\,d^4}{64R^3n} = \frac{12{\cdot}5 \times 10^6 \times {\cdot}232^4}{64 \times {\cdot}45^3 \times 35} = 200 \text{ lbs.}$$

Two such springs satisfy the conditions exactly, provided the shear stress is not excessive.

It is given by—

$$f_s = \frac{16\,WR}{\pi\,d^3} = \frac{16 \times 800 \times {\cdot}45}{\pi \times {\cdot}232^3} = 146000 \text{ lbs. per in.}^2$$

This shear stress is rather high, but if the deflection of the springs were limited to, say, 3·5 ins., then the arrangement would be satisfactory.

The remainder of the work hardly needs comment, it being necessary to see that (1) the skid post can take any bending stresses; (2) that the skid tube is fairly stiff, since the reaction may not pass through the pin at the spring socket, thus causing the skid tube to take bending; (3) that the whole arrangement will be able to withstand a load in the control cable corresponding to a load of 100 lbs. applied at the pilot's foot; (4) that the loads coming on the members of the skid can be safely transmitted to the fuselage.

CHAPTER XIV

CONTROL SURFACES

Ailerons

To determine the intensity of air loading upon any of the control surfaces of an aeroplane with some degree of exactness is not the easiest of matters in the absence of experimental work, and of all the control surfaces the ailerons are probably not the least difficult in this direction.

Some attempt, however, must be made from the standpoint of design, and it is suggested to employ the two following methods in making an estimate of the loading, and then to design the ailerons to the greater of the two loadings obtained from these methods.

(*a*) In ordinary steady flight with the centre of pressure at ·5 of the chord, the ailerons are performing their duty as lifting surfaces, and are probably subjected to the worst conditions of loading they can experience. From the load curve (Fig. 20), the load on the top wing from the wing tip to the inner end of the aileron (*i.e.*, a length of 7 ft.) is 97 × 11·2 = 1088 lbs. Now since the centre of pressure is at ·5 of the chord, and assuming a parabolic distribution of load along the chord, the load taken by one aileron when operating as a lifting surface will be—

$$\frac{2{\cdot}08^2\,[3 \times 5{\cdot}75 - 2 \times 2{\cdot}08]}{5{\cdot}75^3} \times 1088 = 325 \text{ lbs.}$$

the wing and aileron chords being 5·75 ft. and 2·08 ft. respectively.

Fig. 78 shows diagrams illustrating the variation of loading along the chord for C.P. at ·3 and at ·5 times the chord. The area of one aileron = 15 sq. ft., hence the intensity of loading is 21·7 lbs. per square foot.

(*b*) In the second method the ailerons are assumed to be in operation as control surfaces, in which case the loading is dependent upon the maximum load exerted by the pilot upon his control stick. Unless the hand-wheel type is used, the movement of the control stick is lateral when operating the

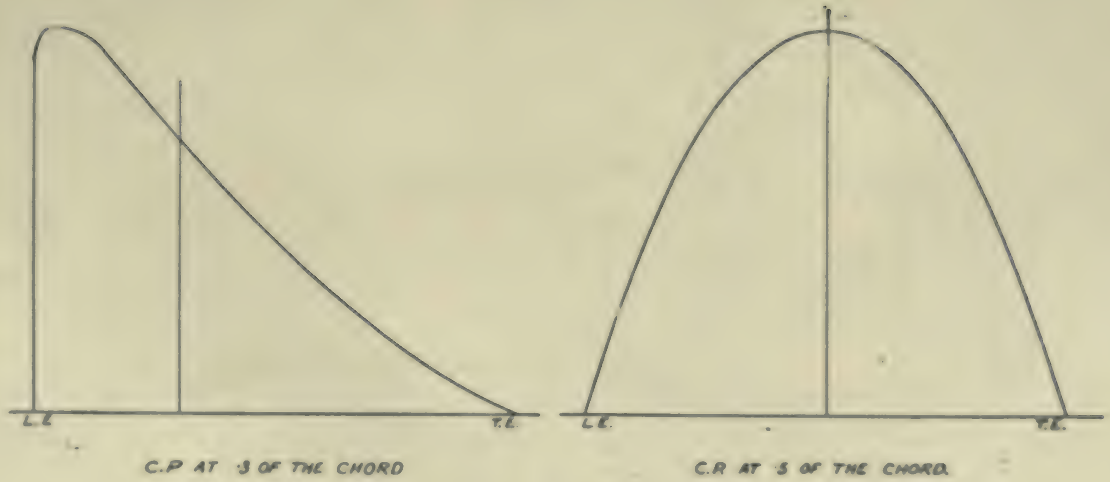


FIG. 78.—DIAGRAM SHOWING VARIATION OF LOADING OVER THE CHORD OF AN AEROFOIL

ailerons, and the maximum load which a pilot can exert in that direction has been assumed to be 50 lbs., so that, knowing the leverage of the controls, it is possible to find the intensity of loading on the ailerons.

In this case the maximum load on the ailerons is 6·42 lbs. per sq. ft., which is less than that previously obtained in (*a*).

In the absence of further knowledge the aileron itself must then be designed to the load of 21·7 lbs. per sq. ft. *i.e.*, according to the intensity of loading when the aileron is operating as a lifting surface.

Aileron Spar

Total load on one aileron = 325 lbs. Assuming that each of the seven ribs and the tip take load corresponding to the area each supports, then the load on ribs 1 to 6 is 46·5 lbs. on each, since the ribs are 12 ins. apart.

The lever rib is made particularly strong in order that it may transmit a fair proportion of the load direct to the cable supporting the aileron. It is assumed that the stringer and trailing edge are strong enough to transmit the whole of the load on the area shown shaded (Fig. 79) to the lever rib without putting torsion in the aileron spar. On this assumption the maximum torsion in the spar will be due to the load on ribs 5 and 6, and the end rib *i.e.*, 2·5 × 46·5 = 116·2 lbs.

Thus torque on spar = 116·2 × 7·75 = 900 lbs. ins., since the centre of pressure is assumed to be at ·33 of the aileron chord.

The resistance of a hollow spar to torsion is given by—

$$2\,A\,f_s t = \text{torque}$$

where A is the area of the outline of the mean section in square inches, t = minimum thickness in inches, and f_s = allowable shear stress in lbs. per square inch.

This rule was first published by Batho in *Engineering* (1917). It represents the greatest advance in our knowledge of torsion problems for half a century, and is, consequently, well worthy of note. It was first used in aeroplane design by H. A. Webb, M.A., at the Royal Aircraft Establishment, Farnborough, in 1917.

Actually it is absolutely true for infinitely thin tubes, whether circular or otherwise, and, according to Webb, is not more than 5% out for sections of finite thickness, such as box spar sections now so popular with aeroplane designers.

We are indebted to Mr. Webb for the following simple proof of the rule, which has not hitherto been published.

The tube need not be circular, nor of uniform thickness. Take any small element P.Q. of tube where the thickness is t, and shear stress f_s; p is the length of the perpendicular from O, the axis of torque, on to the tangent to the tube at p. Of course in the special case when the tube is *circular*, p is the radius.

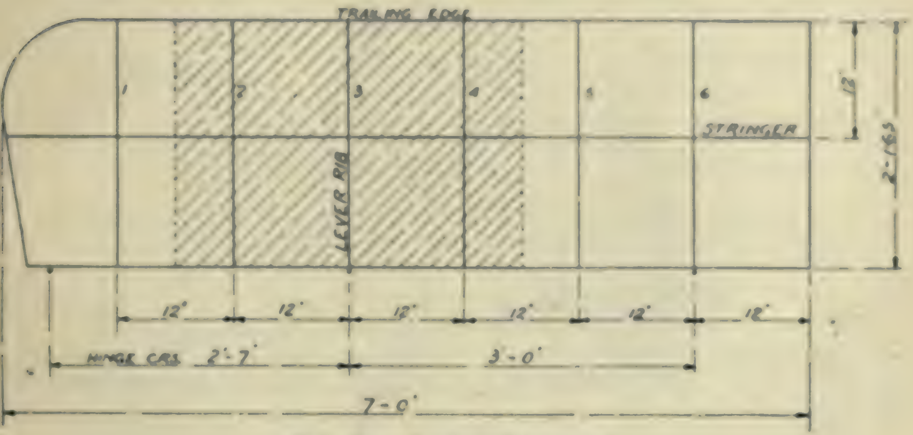


FIG. 79.—PLAN OF AILERON

The thickness t is assumed small compared with p. Now from the equilibrium of the small element it is clear that f_st, which is proportional to the shearing force, is constant at all points of the section.

Then, taking moments about O, torque on whole section

$$= \Sigma\, f_st\, p\, P\, Q$$

$$= f_st\, \Sigma\, 2\, \triangle\, O\, P\, Q$$

$$= 2\, A\, f_st$$

It will be remembered that since f_st is constant at all points, then if the thickness varies, f_s will vary inversely as t. Thus, using the rule in order to obtain maximum stress over the section the minimum thickness must be taken.

Applying this rule to the example, we have, from Fig. 80, which shows the section of the spar—

$$A = 1{\cdot}3 \times 2{\cdot}8 = 3{\cdot}64 \text{ sq. ins.}$$

$$t = {\cdot}3 \text{ in.}$$

The torque taken by the spar is 900 lbs. ins.

Then $900 = 2 \times 3{\cdot}64 \times {\cdot}3 f_s$

From which

$$f_s = \frac{900}{2 \times {\cdot}3 \times 3{\cdot}64} = 410 \text{ lbs. per sq in.}$$

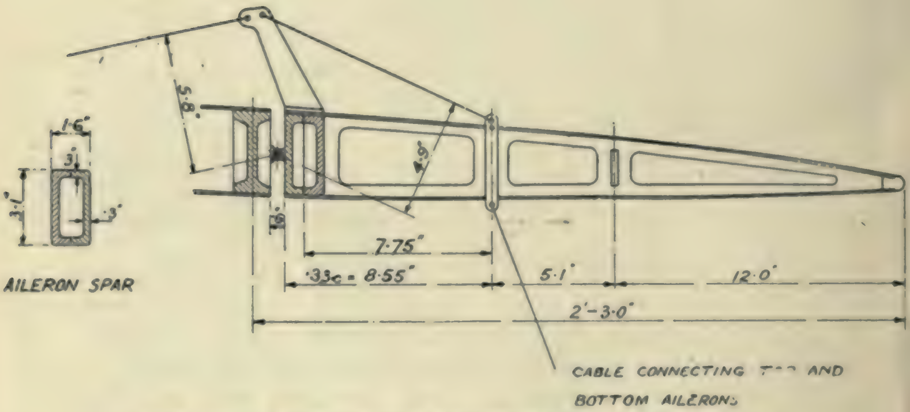


Fig. 80.—Section of Aileron Showing Spar and Control Fittings

The allowable stress for good spruce in torsion is somewhere in the neighbourhood of 400 lbs. per sq. in., and it is quite possible that this figure represents a good mean between the various results of tests. The spar is therefore just about up to strength in torsion.

The bending of the aileron spar must also be considered, since the distance between any two of the three hinges is 36 ins.

$$\text{Uniform load on spar} = \frac{325}{6{\cdot}7} = 48{\cdot}5 \text{ lbs. per ft.}$$

Since the unsupported length of spar = 3·0 ft., then bending-moment at middle of span, if ends be taken as free, is

$$\frac{48\cdot5 \times 3^2}{8} = 54\cdot5 \text{ lbs. ft.}$$

$$I \text{ of spar section} = \frac{BD^3 - bd^3}{12} = \frac{1\cdot6 \times 3\cdot1^3 - 1 \times 2\cdot5^3}{12}$$

$$= \frac{47\cdot5 - 15\cdot6}{12} = \frac{31\cdot9}{12} = 2\cdot66$$

$$Z \text{ of section} = \frac{2\cdot66}{1\cdot55} = 1\cdot72$$

$$\text{Bending stress in spar} = \frac{54\cdot5 \times 12}{1\cdot72} = 380 \text{ lbs. per sq. in.}$$

hence, the spar is all right in bending.

The load on the hinges is easily obtained from the load per foot length of the spar, but the reactions ought really to be corrected for fixing moments on the spar at the hinges.

Plenty of clearance should be allowed at the hinges, which, if possible, should be limited to three for easier alignment.

The attachment of the ribs to the spar should be made quite definite, particularly so in the case of the lever rib, which is heavily loaded. Usually wood screws are used for this purpose, reinforced by securely glueing the joint. An estimation of the strength of such a connection is difficult except by comparative tests.

The strength of the lever rib can, however, be determined by equating the bending-moment to the moment of resistance of the section.

Suitable sections for the aileron ribs in this example are shown in Fig. 81. If the stringer is to be efficient in its transfer of load from rib to rib it should be made as deep as the section will allow and thick enough for stiffness.

Bottom Aileron Lever

This case is also severe on the lever on the bottom aileron, for the load in the cable from the quadrant on the control stick to the bottom aileron lever is fairly large ; its amount is shown to be 910 lbs. (*see* Chap. XVI).

The load in this cable which connects the two bottom ailerons and maintains them in their correct relative position to the wings when flying, is entirely independent of any control stick load.

The true load in cable connecting top and bottom ailerons is $\frac{325 \times 60{\cdot}8}{56} = 353$ lbs., allowing for the inclination of the cable to the vertical. The cable attachment to the lever rib is assumed to be at the point where C.P. acts. This load, and the load 140 lbs. on the portion of bottom aileron, shown shaded, must be transmitted through the lever rib to the lever which is thus put into compression due to this load.

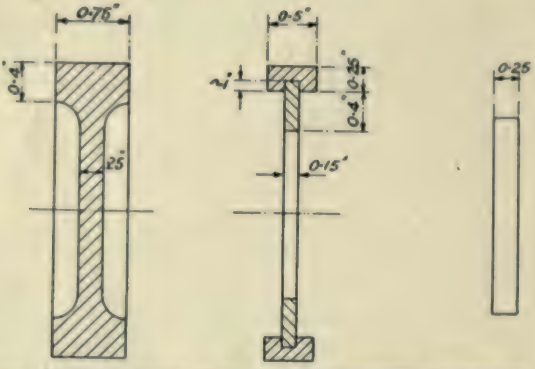


FIG. 81—AILERON RIB AND STRINGER SECTIONS

There is a certain amount of torsion of the bottom aileron spar exactly as in the top aileron, and this is taken as bending on the lever. The compression load in the lever can easily be found by drawing the triangle of forces.

ELEVATORS

The principle of much of the work already done on the ailerons and to be done on rudder may be applied directly to the design of the elevator, hence it is proposed merely to indicate how to estimate the probable maximum load intensity and then to refer the reader to the portions mentioned above.

In our example, we shall see, in working through the elevator controls, that the load on one elevator, due to a force of 100 lbs. at the pilot's hand, is 180 lbs. Now, the area of one elevator would probably be about 9 sq. ft., since the area of tail plane and elevators = 50 sq. ft. (*i.e.*, elevators = 18 sq. ft.).

Thus the load intensity works out to be about 20 lbs. per square foot.

If now we consider the loading from another standpoint, (*i.e.*, taking the maximum load, 1500 lbs., already found for the tail as being uniform over the tail plane), we shall get a loading of $\frac{1500}{50} = 30$ lbs. per square foot.

Obviously, if the elevators be designed to this higher figure, they may become unnecessarily heavy, and being, as they are, so far away from the centre of gravity of the machine, this would be undesirable.

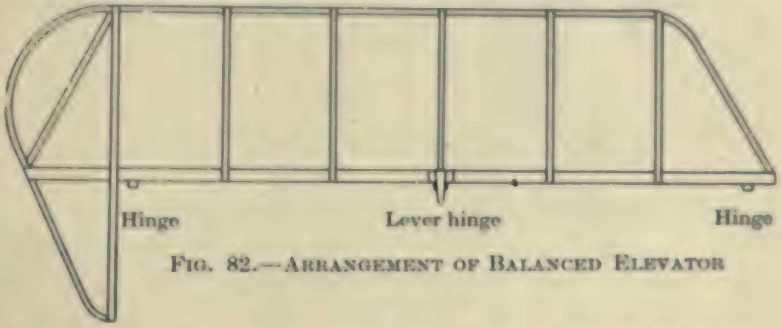


Fig. 82.—Arrangement of Balanced Elevator

It is not advisable, however, to design elevators to a load of less than 20 lbs. per square foot.

The remainder of the work in the design of the elevator spar to resist torsion, and of ribs to resist bending and shear, etc., is very similar in character to that on the design of ailerons which has already been done in considerable detail, and is therefore not necessary to repeat here.

Elevator spars are usually made of spruce very much like aileron spars, but in some cases steel tubular spars are employed. In such cases, the whole elevator is generally made up of steel throughout, and can often be made lighter than wooden designs, beides having a probable longer life.

Balanced elevators are often fitted ; this gives the pilot less fatigue and possibly greater manoeuvrability, but too much balancing is not advisable, since it would " take charge " after relatively small angular movements. All the load on the elevators, including the balanced portion, must be taken by the rear tail plane spar.

Fig. 82 shows an example of balancing.

CHAPTER XV

CONTROL SURFACES—(*Continued*)

RUDDER

RUDDERS are now generally constructed of steel tube for the outer boundary with spruce wooden ribs set and attached to steel sockets, which in turn are either brazed or sweated to the rudder post and trailing edge. Frequently, however, metal ribs also are designed and suitably fixed direct to the rudder post and trailing edge. Fig. 83 shows an arrangement of rudder. One of the ribs is generally made of somewhat larger section, and is known as the lever rib.

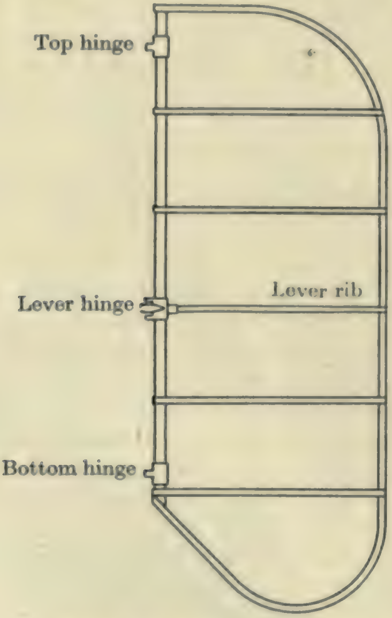


FIG. 83.—ARRANGEMENT OF RUDDER

The usual assumption is that any one rib will support the load on the area bounded by lines drawn midway between it and the ribs adjacent to it; furthermore, the position of the centre of pressure may be assumed to be at ·33 of the rudder chord. The area of the rudder has already been fixed at 9 sq. ft., and the chord of the rudder at 2 ft., hence the centre of pressure will be at 8 ins. from the centre line of the rudder post.

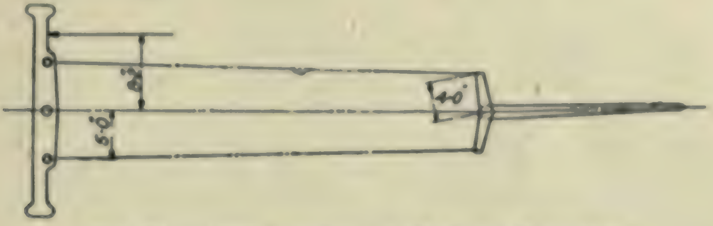


Fig. 84.—Arrangement of Rudder Controls

The load on the rudder due to a force of 100 lbs. at the pilot's foot will therefore be (*see* Fig. 84)—

$$\frac{100 \times 8{\cdot}2}{5} \times \frac{4}{8} = 82 \text{ lbs.}$$

i.e., about 9 lbs. per square foot.

If this loading had been greater than 15 lbs. per square foot it would have been adopted as the loading for the design of the rudder, but since it is less it is usually ignored, and the higher figure of 15 lbs. per square foot chosen.

Thus we may generalize by saying that intensity of design load for the rudder is chosen as the greater of the two loads, *i.e.*, 15 lbs. per square foot or the load due to a force of 100 lbs. at the pilot's foot.

In the example the total load on the rudder will therefore be 9 × 15 = 135 lbs., and is assumed to operate at the position of the centre of pressure which has been taken at 8 ins. from the centre of the rudder post.

Rib Design.—Each rib must be designed to take the bending-moment due to the load it carries. It is required to transmit the load to the rudder post, and it must therefore be well fixed to the post and to the trailing edge. The trailing edge is often made of steel tube about ⅜ in. diameter, and of 20 or 22 gauge thickness, which can easily be bent to suit the shape of rudders in general; it is not, however, relied upon as having any strength.

The rudder post is made from steel tube of suitable diameter and gauge to withstand the torsion induced by the moment of the air forces on the rudder.

Thus in the example the total torque on the rudder post is $135 \times 8 = 1080$ lbs. ins., but the whole of this torque is not taken by one section, since the lever is near to the centre of the rudder post.

The procedure is almost identical with that of the design of ailerons, a reference to which will indicate how the load is to be divided between the various ribs, and the proportion of the torque taken by any one section of the rudder post. It is probably superficial at this stage to do the whole of the work in detail since it is almost a repetition of the work done in the design of ailerons as already described.

When the maximum torque on the rudder post has been determined, it may be equated to the torsional resistance of the steel tube.

Thus $T = \frac{2qJ}{D}$

where T = maximum torque on any section
J = polar moment of inertia
D = outer diameter of tube
q = allowable shear stress in the tube material.

A certain amount of bending of this tube is also likely, but it must be left to the discretion of the draughtsman or designer in any particular example to decide whether or no it is worth accounting for. If in a particular example it is estimated to be large, then the bending stress must be combined with the shear stress due to torque by the ordinary methods generally used.

LOADS ON HINGES.—Before leaving this portion of the work it is advisable to estimate the load on the hinges, since they will be required when designing the fin post to which the rudder is hinged.

Not less than three hinges should be used, and sufficient clearance ought to be allowed for purposes of interchangeability.

One of the hinges should be set as near as possible, if not actually at, the position of the lever on the rudder post, for the reaction is probably a maximum at that point. The loads at the other hinges are proportional to the area of rudder they may be assumed to support, and there will be no difficulty in estimating them.

FINS

The design of fins is not widely different from rudder and elevator design, but the difficulty of knowing exactly what is the greatest likely load intensity on the fins is one of the designer's uncertain quantities. Sufficient experimental work has not yet been completed so as to enable designers to fix upon load intensities which may be relied upon with absoluteness. It was, therefore, necessary, in early designs, to make discreet assumptions in this respect, and then to see how the fin structure was able to withstand the loads occurring in various attitudes of flight. The experience thus gained has enabled more recent designers to use a figure of loading which may be relied upon to produce a reasonably strong structure. This figure is in the neighbourhood of 10 lbs. per square foot. Fins have been set at different places on aeroplanes as the science has progressed, and in the very early days they were often set well in the front of aeroplanes, but are now almost invariably set at the rear of the machine, where better stabilizing effect is obtained.

Very frequently, a smaller portion of the whole fin area is set beneath, and the major portion above, the fuselage.

In the example, the total area of fin surface is 7 sq. ft., 5 of which is above, and 2 below, the fuselage (*see* Fig. 85).

The fins are attached to the fuselage structure by steel or duralumin clips set at intervals along the wooden rib, which is in direct contact with the fuselage.

Both top and bottom fins are bounded on their leading edges by small diameter tubing, about ⅜ in. diameter, of, say, 22 S.W.G., and bounded on their after edge by the fin post, which is telescoped into the sockets which form part of the rear end of the fuselage.

The fin post would be made of steel tubing of suitable diameter and gauge, and would be made in two pieces, in order that the upper and lower fins may be detached separately.

The streamline wire bracing, which supports the tail plane, is attached to the upper and lower extreme ends of the fin post.

Metal or wooden ribs are placed in the fins in suitable positions to stiffen the structure and support the doped linen covering.

Regarding the strength of the fins, the leading edge is not supposed capable of contributing to any extent worth estimating, and it is therefore necessary to design the ribs as cantilevers supported at the fin post where strong metal clips are necessary.

The fin or stern post is really the important member of the fin, because not only does it help to support the fin itself, but also the tail plane, and, furthermore, the rudder is also hinged to the fin post. Side load due to the air forces acting directly on to the fin may be assumed to be uniform along the length of fin post affected, while the loads due to air forces on the rudder will be transferred to the fin post as concentrated loads at the hinges.

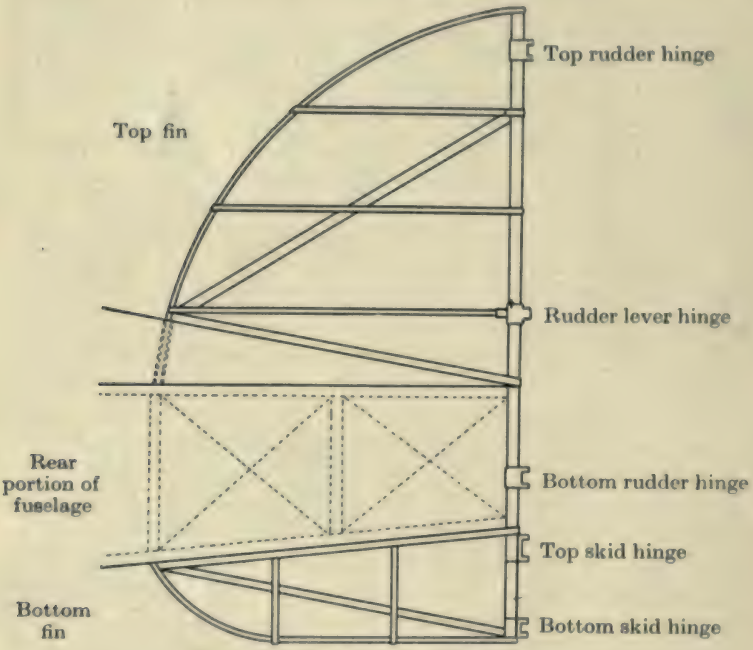


FIG. 85.—ARRANGEMENT OF FINS

The fin post may be assumed fixed at its point of attachment to the fuselage, and must withstand the bending-moment due to the moment of the total air force on the upper fin.

It may be argued that this is somewhat unfair and rather severe on the fin post, since the bracing wires from the tail plane are attached to both upper and lower portions of the fin post.

An alternative to take is, therefore, on the assumption that the upper and lower portions of fin post, and the tail plane spars may mutually help in resisting this lateral moment due to load on the fins, by their combined bending resistances.

This is probably the more correct point of view, because the streamline wire bracing is capable of transmitting load between the upper and lower portions of fin post. Incidentally, however, it means that end load is introduced in the tail plane spar and in the upper and lower portions of fin post, and, furthermore, that bending in the rear tail plane spar, as well as in upper and lower portions of fin post, is also introduced. If the former assumption is made in estimating the requisite size of the fin post, an ample margin of safety will be assured.

It is assumed that both upper and lower fins sustain the same intensity of loading simultaneously, hence the moment of the upper fin load will partially counterbalance that of the lower fin load through the medium of the streamline bracing wires.

The resultant moment which has to be resisted by the structure as a whole is, therefore, the difference between the moments of the air forces on the upper and lower fins. The moment of load on upper fin post = load intensity on upper fin and rudder × area of upper fin and rudder × distance of centre of pressure of fin and rudder from centre line of fuselage = M.

Similarly, moment on lower fin post = load intensity on lower fin × area of fin × distance of its centre of pressure from centre line of fuselage = m.

These being in opposition, the resultant moment is therefore

$$M - m = M_R$$

and this must be resisted by the upper and lower portions of fin post and the tail plane spar together.

The general procedure in the estimation of the stresses in the fin structure is therefore—

. (1) Find the uniform load on the upper fin post due to lateral air forces on the upper fin and concentrated loads due to air forces on rudder, the intensities of the air forces on fin and rudder being taken at 10 and 15 lbs. per square foot respectively. The concentrated loads at rudder hinges are readily obtained from the rudder calculations.

(2) Estimate the uniform load on the lower fin post due to lateral load on lower fin of 10 lbs. per square foot.

(3) Calculate the moments of these air forces on rudder and upper and lower fins, and determine the resultant moment.

(4) Find the end load in the fin post due to the forces in the streamline bracing wires.

The problem is now resolved to that of members subjected to bending-moments as cantilevers, with end load. It is, of course, advisable that the whole of the moment be absorbed by the fin post, rather than the spar being relied upon to take any portion, since it may be stressed by loads on the tail simultaneously. In practice, no doubt each member will take its share of the moment according to its ability to resist it.

Assume a suitable size of tube for the fin post and find from Table 31 the values of its sectional area and modulus. Then, if the spar is not to be relied upon to take any bending due to load on fins, the tube chosen must, of course, withstand it.

The stress is composed of a portion due to bending and the remainder due to end load direct.

$$\text{Stress due to bending} = \frac{BM}{Z} = f_1$$

$$\text{Stress due to end load} = \frac{W}{A} = f_2$$

$$\text{Total stress} = f_1 + f_2$$

and this must be less than the allowable stress for the material of which the tube is made.

If the deflection is likely to be excessive, and it is thought desirable to check it, the bending-moment diagram must be integrated graphically twice to produce the deflection diagram, from which such deflection may be measured to scale.

The bottom fin post is probably subjected to its worst condition of loading by the skid load which occurs on landing.

The leading edge of the lower fin is thus in tension unless a special internal member is inserted in the fin to take such tension. Bending in the lower fin post is caused by the skid hinge reactions and if the tube is made to withstand this bending, it will generally be strong enough to withstand any other case of loading on the lower fin.

CHAPTER XVI

FLYING CONTROLS

The term "flying controls," is understood to include the mechanism and cables controlling the ailerons, elevators, rudder, and also the tail plane, where the latter is adjustable from the pilot's seat, as it sometimes is in modern aeroplanes.

Usually the ailerons and elevators are operated by the control column, by a movement in two directions at right angles.

For instance, the fore and aft movement of the control column, or "joy-stick" as it is often termed, lowers and raises the elevators respectively, the machine tending to nose dive or stall, while the side-to-side movement operates the ailerons; a movement of the column to starboard raising the starboard aileron and lowering the port aileron simultaneously, thus the starboard wing loses lift and is depressed, while the port wing is raised.

The rudder is usually controlled by a swivelling bar known as a "rudder bar," operated by the feet in such a manner that a forward movement of the right foot swings the rudder, which accomplishes a right-hand turn of the machine, and *vice versa*.

For different sizes and types of aeroplanes, the arrangement of controls must necessarily vary, although the general principle remains the same, and hence each separate case needs some slight difference of treatment. The best example which might be cited as a general case is probably the case of a biplane having ailerons on both top and bottom planes, having a monoplane tail and a single rudder, suitable for an aeroplane up to 5000 lbs. weight.

Aileron Controls

Considering first the case of the ailerons, it will be remembered that of the two methods of estimating the loading thereon, the maximum is obtained when the ailerons are merely considered as part of the lifting surface of the whole plane as against when they are being operated as control surfaces. Reference to the work already done on strength of ailerons (Chap. XIV) will confirm this. Nevertheless, the cables connecting the control quadrant to the bottom ailerons and the cable connecting the top and bottom ailerons will have to withstand loads due to the maximum lift force on the ailerons, because these cables maintain them in position as lifting surfaces.

Several arrangements of aileron cables might be considered, but it is thought to be unnecessary to show detailed calculations for every such conceivable scheme, since there is considerable similarity between them, and it is enough to show how it may be done for one popular arrangement. In this arrangement (Fig. 86) the two upper ailerons are directly connected by a cable which passes along the top front spar inside the plane, each end passing round a pulley at right angles, after which it passes out through the upper fabric surface of the plane to the respective top aileron lever, which latter is fixed to the aileron spar and projects above the plane. There is also a cable connection between the aileron lever and a fitting on one of the aileron ribs known as the lever rib. The top and bottom ailerons are directly connected by a single cable, whose ends are attached to metal fittings on the lever ribs, the lever rib being left solid, or with very little spindling for purposes of strength. On the underside of the bottom aileron spar is attached a lever which projects below the bottom plane in exactly the same way that the lever on the top aileron was fixed above the upper plane.

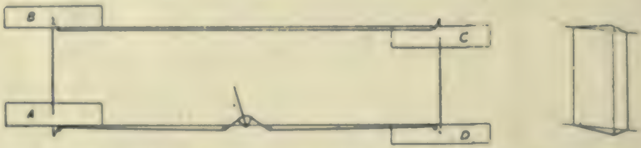


FIG. 86.—ARRANGEMENT OF AILERON CONTROLS

A control cable, being fixed to the quadrant at the pivot of the control stick, passes along the bottom front spar inside the plane, then passes round a pulley at right angles and emerges through the bottom fabric surface of the lower plane to this lever. A cable connection from the lever to the lever rib exists in order to minimize the bending on the lever rib. The cable arrangements described above are the same, of course, on both port and starboard sides of the aeroplane.

It will now be clear that the cable connecting the two upper ailerons only takes load when the ailerons are operated as control surfaces or in the down load case, of which the latter produces the greater load. It was shown in Chapter XIV that the load on one top aileron is 325 lbs. when the centre of pressure on the planes is at ·5 of the chord, and this is taken wholly by the inter-aileron cable. In order to find the true load in this cable, which is not in the vertical plane, this load of 325 lbs. must be increased in the ratio of the true length to the vertical distance, namely, 60·8 to 56 in the example. The true load in this cable is therefore—

$$325 \times \frac{60{\cdot}8}{56} = 353 \text{ lbs.}$$

A 10 cwt. cable would, however, be used.

Load in Cable from Bottom Aileron to the Control Stick

The maximum load is that due to the lift forces on the top and bottom ailerons. Load on one top and one bottom aileron, allowing for the efficiency of the latter, $= 325 + {\cdot}85 \times 325 = 601$ lbs.

Then, taking moments of this force about the hinge point, the load in the cable considered is—

$$\frac{601 \times 8{\cdot}8}{5{\cdot}8} = 910 \text{ lbs.} \quad (\textit{See } \text{Fig. 80.})$$

The strength of cable required for this load would be a 10 cwt. size.

It will be noted that these loads are effected without any force being exerted on the control stick by the pilot.

Load in Top Aileron Connecting Cable

We shall first consider the load in this cable due to the force exerted by the pilot at the control stick; consequently, we must commence there in order to find this load. It has been found that a pilot cannot exert more than 50 lbs. laterally on the control stick when operating the ailerons. Fig. 87 shows diagrammatically the arrangement of the control stick controlling the four ailerons referred to as A, B, C, D. It is imagined that the control stick is operated so that the ailerons C and D are pulled downward, and the reasonable assumption is also made that the loading due to air lift forces is, in the case of ailerons down, double that for ailerons up, as in C and D and A and B respectively. The area of each aileron in example is 15 sq. ft. Let the loading per square foot on A and B be x lbs. per square foot, then the loading on C and D will be $2x$ lbs. per square foot. The centre of pressure of the air forces is approximately at ·33 of the aileron chord, *i.e.*, at a distance 8·8 in. from the hinge (Fig. 80). Total load on aileron A = area × load intensity = $15 \times x = 15x$ lbs.

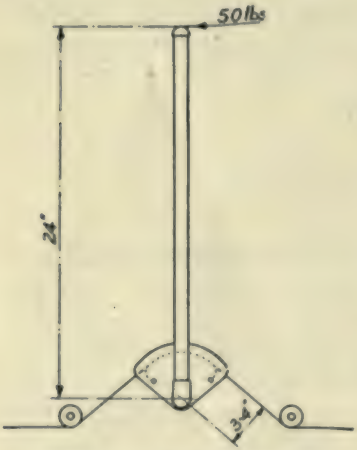


FIG. 87.—CONTROL STICK

Total load on aileron B = $15x$ lbs. also, the load on A being transferred through the connecting cable from A to B, which cable is attached to the ailerons at approximately the centre of pressure of the load on the ailerons.

In order to find the load due to ailerons A and B in the cable connecting the two top ailerons B and C, moments about the hinge points must be taken, thus—

$$\frac{2 \times 15x \times 8{\cdot}8}{5{\cdot}8} = 45{\cdot}5x \text{ lbs.} \qquad (\textit{See}\ \text{Fig. 80.})$$

The load on the aileron C is $2 \times 15x$ lbs., and the load on the aileron D is $2 \times 15x$ lbs., hence the load in the control cable from the control stick to the aileron D is obtained by taking moments about the hinge point, *i.e.*—

$$\frac{(2 \times 15x + 2 \times 30x)\ 8{\cdot}8}{5{\cdot}8} = \underline{136{\cdot}5x \text{ lbs.}}$$

To find the magnitude of loading x, we need to find the load in the control cable just considered due to a lateral load at the pilot's hand of 50 lbs., and then equate to the quantity above. Thus the load in the cable is—

$$\frac{50 \times 24}{3{\cdot}4} = 353 \text{ lbs.}$$

hence

$$x = \frac{353}{136{\cdot}5} = 2{\cdot}59 \text{ lbs. per sq. ft.}$$

Thus, the maximum load in the cable connecting the two top ailerons on this consideration = 45·5 × 2·59 = 118 lbs. It is probable, however, that this cable will have to withstand a greater load in the case of down load on the wings. This latter load can easily be estimated by taking three-fifths of the load found in the cable connecting the control stick to the bottom aileron when the ailerons are performing their functions as lifting surfaces, the load being $\frac{3}{5} \times 910 = 546$ lbs.

This is obviously greater than that obtained above and determines the size of cable required. A 10 cwt. cable would obviously be quite strong enough for this load.

Other Arrangements of Aileron Cables

Another arrangement is that in which a light strut connects the upper and lower ailerons; and the cable from the quadrant, (which latter, in this case, is at the rear end of a rocking shaft passing beneath the pilot's seat), passes along behind the bottom rear spar up to a point just short of where the lower aileron commences, when it passes round a pulley and is connected to the main rib of the upper aileron at the top of the aileron strut. From the bottom end of this strut, another cable is connected to the upper plane, where it passes round a pulley and thence along behind the rear spar of the upper

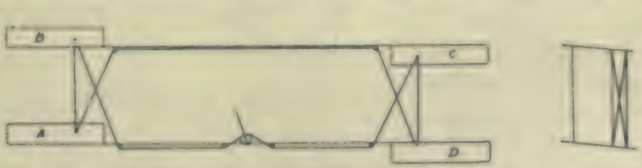


FIG. 88.—ALTERNATIVE ARRANGEMENT OF AILERON CONTROLS

plane and down to the bottom end of the aileron strut on the opposite side of the machine. Figs. 88 and 89 will show the arrangement above described.

This would be quite satisfactory for a comparatively small machine, where the steepness of the cables makes them more efficient, than would be the case in the larger machine, in which the cables would deviate from the vertical still more, and thus produce slackness, which could not be taken up.

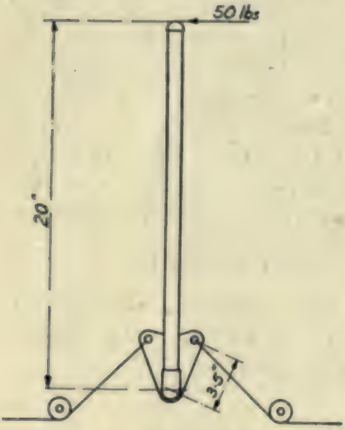


Fig. 89.—Control Stick

This system was used with success on the Austin "Whippet" and the R.A.F. two seaters B.E.2E. and R.E. 8.

The control stick has not been stressed here, for it will have to withstand a greater stress when operating the elevator controls where the movement of the stick is fore and aft, and the load at the pilot's hand is 100 lbs.

It is essential that the pulleys should be set at the exact angle, so that there is no tendency for the cable to leave the pulleys. All pulleys should be mounted on ball bearings, standard sizes of which can be utilized for all the ordinary sizes of pulleys in use.

Cable guides should always be fitted, for then the danger of a cable tending to jump the pulley at entry is diminished. In order to have reasonable sensitiveness of control, the angular movement of the aileron per 3 ins. movement of the pilot's hand should have an average value of about 4 degrees.

RUDDER CONTROLS

As much as can be done on the rudder controls is generally more or less straightforward, and it is only very rarely that arrangements of rudder controls vary from what has become acknowledged as standard practice. Consequently it is proposed to show, as an example, how to estimate the strengths of the various parts of a typical system of rudder controls as shown in Fig. 84.

Referring to the rudder bar itself, it, of course, is operated by the pilot's feet, and the control cables are attached to it either at the extreme ends or on the inner side of the pilot's feet, *i.e.*, nearer to the pivot upon which the rudder bar is capable of swivelling. The advantage of attaching the cables to points on the rudder bar within the pilot's feet is that greater leverage is obtained compared to the alternative case (Fig. 90), where the cables are attached to the ends of the bar. More control is, therefore, exercised over the rudder for a given load at the pilot's foot. Over-sensitiveness of control is a point which needs consideration in all flying controls or they may prove troublesome in operation.

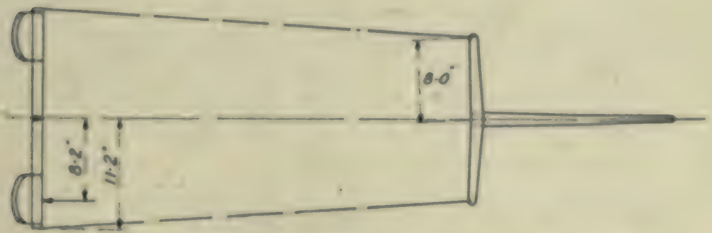


FIG. 90.—ALTERNATIVE ARRANGEMENT OF RUDDER CONTROLS

The movement of the rudder must not be excessive compared to the movement of the pilot's foot. For example, for a movement of 3 ins. of the pilot's foot the angular movement of the rudder may be taken as 21 degrees, and this may be accepted as a good average value. On the other hand the control must not be under sensitive or it may prove disastrous in case of emergency.

These matters may be varied quite easily by modifying the length of the rudder lever, and also the relative leverage of the pilot's foot and rudder cable at its point of attachment to the rudder bar.

In performing the calculations required, the reader is referred to Fig. 84, where the distances of the point of attachment of the cable, and that of the pilot's foot from the rudder bar pivot, are respectively 5 ins. and 8·2 ins.

Assuming, as is already suggested, that the maximum load applied by the pilot's foot = 100 lbs., then the load in the cable is found to be—

$$\frac{100 \times 8{\cdot}2}{5} = 164 \text{ lbs.}$$

It is very unlikely, however, that this cable will be less than 10 cwt. strength, but it ought to be noted that if in another machine the load were greater, it does not follow that a corresponding increase in size would be made.

Rudder bars vary a great deal in their design, but one of the simplest and yet satisfactory forms is that in which a single steel tube with suitable fittings is used.

Let it then be assumed that the rudder bar is essentially a steel tube $1\frac{1}{8}$ in. diameter, of thickness, say, 20 S.W. gauge, and of Air Ministry specification, say, T2; Z for such a tube will be found to be $0{\cdot}0326$ in.^{3} (*see* Table 31 in Appendix). The maximum bending-moment will be inward from the point of attachment of the cable to the rudder bar, namely,

$$100 \times 3{\cdot}2 = 320 \text{ lbs. ins.}$$

The maximum stress will, therefore, be—

$$\frac{320}{0{\cdot}0326} = 9800 \text{ lbs. per sq. ins.}$$

which is well within the safe limit of stress for this material.

Another form of rudder bar is made from two pressings of suitable form riveted together with perhaps a core of wood. The pressings may be of aluminium or duralumin, or even steel. Other forms are also sometimes designed.

Bearing of Rudder Bar Support

The total maximum load ordinarily likely to be imposed on the rudder bar support is the difference between the pull in the cable and the load at the pilot's foot (*i.e.*, 164 − 100 = 64 lbs.)

This load of 64 lbs. may easily be accounted for by making the pivot pin of suitable diameter to withstand any possible bending-moment and to allow sufficient bearing surface between the pin and the bracket supporting the rudder bar. This bracket must be secured to flooring and well attached to the fuselage of the machine.

Angular Movement of Rudder

The angular movement of the rudder will now be determined by the length of the rudder lever, which is usually made of pressed steel plate, welded together and securely attached to the rudder post, which is generally made of circular steel tube.

In this example the length of rudder lever has been chosen as 4 ins., that is, the point of attachment of the cable to the lever is at 4 ins. radius.

Continuing the example, let there be a movement of 3 ins. of the pilot's foot, then the point of attachment of the cable will move $\frac{3 \times 5}{8 \cdot 2}$ ins., and the angular movement of the rudder will be such that

$$\tan \theta = \frac{3 \times 5}{4 \times 8 \cdot 2} = 0 \cdot 454$$

hence $\theta = 24\frac{1}{2}^{\circ}$ approx.

Often the rudder lever is attached to a point on a very strong rudder rib, in which case very little, if any, bending-moment occurs in the lever. If, however, no such cable were present, then the lever must be made strong enough to withstand the bending which would then be present.

ELEVATOR CONTROLS

The elevators are operated from the control stick by a fore and aft movement in such a way that a forward movement has the effect of lowering the elevators, and *vice versa*.

Regarding the maximum probable load taken on the elevator controls, it is reasonable to estimate the load as being dependent upon the maximum force which the pilot is capable of exerting at the end of the control column. It has been found that this force is about 100 lbs., and it is proposed to calculate the loads and stresses in the controls due to such a force. The load in the cables from the control column to the two elevators, assuming the load at the pilot's hand to be 100 lbs., will be—
$\frac{100 \times 24}{5 \cdot 1} = 470$ lbs. if the cable is at right angles to the control column, (*i.e.*, 235 lbs. in each cable, *see* Fig. 91).

The position of the lever on the elevator spar is as illustrated, and taking moments of the force in the cable and the whole load

W on one elevator (assumed to be acting at ·3 of the chord) about the hinge centres, we have, neglecting balance portion—

$$W \times 5{\cdot}37 \text{ ins.} = 235 \times 3{\cdot}8$$

hence $\quad W = 166$ lbs.

the chord of elevator being 17·9 ins.

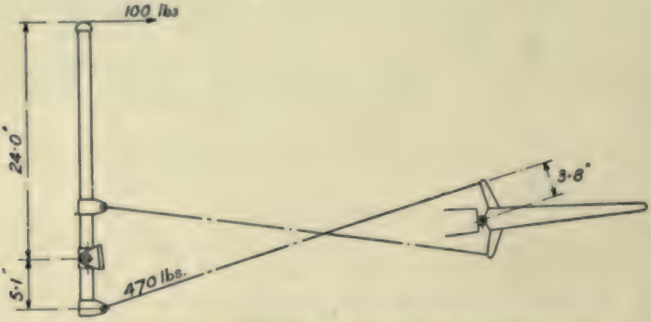


FIG. 91.—ARRANGEMENT OF ELEVATOR CONTROLS

What has already been said as to sensitiveness of control on the rudder applies equally well to the case of elevators,

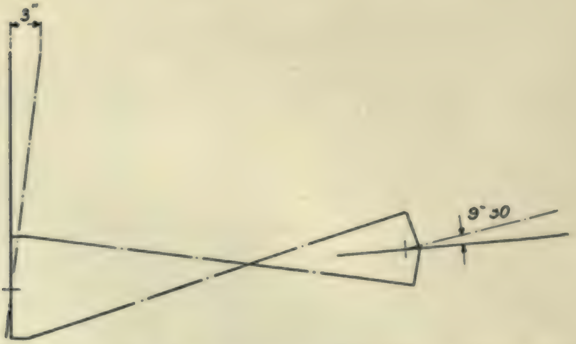


FIG. 92.—DIAGRAM SHOWING ANGULAR MOVEMENT OF ELEVATOR PER 3 IN. MOVEMENT OF STICK

and the happy medium between sluggishness and over-sensitiveness is the ideal condition. In the example (Fig. 92) the angular movement of the elevators corresponding to a

3 in. movement of the pilot's hand will be found to be nearly 10 degrees, and this may be accepted as a good average value.

Thus, a movement of 3 ins. of the pilot's hand produces a movement of the cable of—

$$\frac{3 \times 5{\cdot}1}{24} = {\cdot}64 \text{ ins.},$$

hence the angular movement θ of elevator is such that

$$\tan \theta = \frac{{\cdot}64}{3{\cdot}8} = {\cdot}168,$$

therefore $\theta = 9{\cdot}5$ degrees.

Although the load in one cable is but 235 lbs., yet a 10 cwt. cable would be inserted so as to have no fear of serious stretching.

Of the detail design of the elevator controls, the following are important—

CONTROL COLUMN

This is subjected to a greater force in this case than when considered for aileron controls, the force at the pilot's hand in this case being 100 lbs., whereas it was only 50 lbs. when considered in connection with aileron control. It may be asked why not take the case where both aileron and elevator controls are exercised simultaneously. The reason is that when both are operated together it is hardly possible for the pilot to exercise maximum force in both directions at once, and so the maximum case is probably that now considered.

Bending of Control Column

Sometimes the elevator cables are attached to bloater levers, set just outside the fuselage, being fixed to a cross-shaft which carries the control stick. In that case, the maximum bending-moment would be $F \times d$,

where F = force at pilot's hand

d = distance from pivot where force is applied.

Again, the elevator control cables may be attached directly to the control column itself, in which case the maximum bending-moment would still be at the pivot when the lower elevator cable is taking load (*see* Fig. 91).

In our example, the distance $d = 24{\cdot}0$ ins. and $F = 100$ lbs., hence the maximum $B\,M = 2400$ lbs. ins.

The material of the control column may be steel tube or Delta metal tube having yield stress of 40000 lbs. per sq. in.

Let the tube be 1⅛ outside diameter, and, say, 16 gauge thick.

Z of such a tube in bending would be ·0536 in.^{3} (*see* Table 31 in Appendix).

The maximum stress, therefore, would be—

$$\frac{2400}{\cdot 0536} = 45000 \text{ lbs. per sq. in.}$$

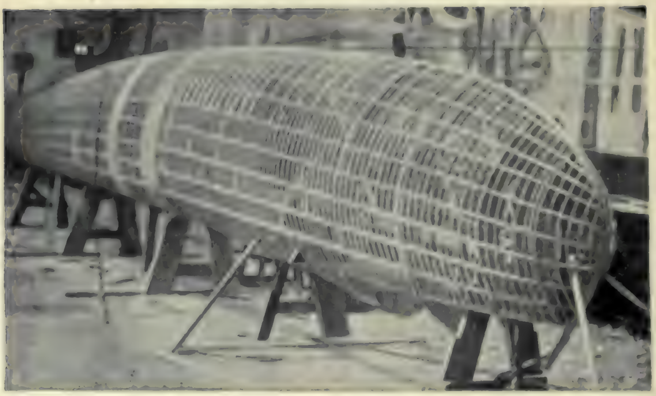
This figure, as it stands, is excessive, but in all probability the tube would be strengthened at that point by sockets, in which case the value of Z would be increased to such an extent that the maximum stress would probably be nicely within 40000 lbs. per square inch.

The bearing on the pivot will be 100 + 470 = 570 lbs., and the pivot pin must be designed to take that force in shear, bending, and bearing.

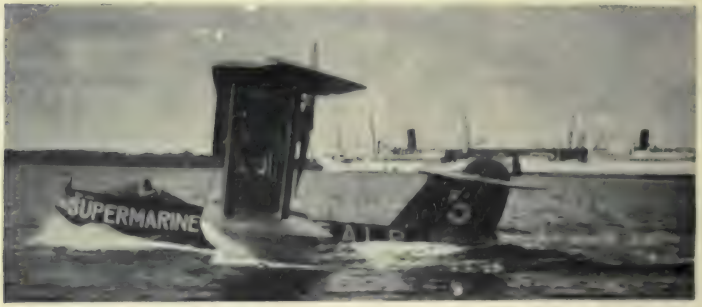
The fixing of the bracket supporting the control stick must be carefully and securely made to the flooring by a suitable size and number of bolts.

The cables from the control column to the elevator should be given as free a path as is possible, and too many bends should be avoided. Cable guides must be inserted wherever necessary at places where slight deviations from the straight line occur. These latter are best made of ordinary compressed vegetable fibre.

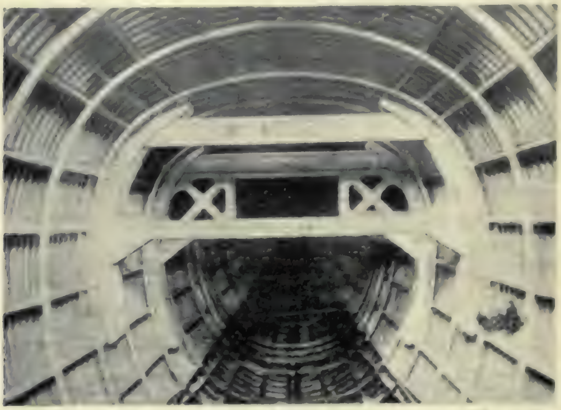
PLATE 7



SUPERMARINE FLYING BOAT HULL—SKELETON FRAME



SUPERMARINE SINGLE-SEATER FLYING BOAT "SEA LION"



SUPERMARINE FLYING BOAT HULL—INTERIOR VIEW WHEN COVERED

CHAPTER XVII

DETAIL DESIGN

To under-estimate the importance of the design of details of most engineering structures is decidedly a shortsighted, if not a dangerous, proceeding, and this is probably more true of aeroplane structural design than of any other branch of engineering. By due consideration of detail, the efficiency of the part may be increased, the method of manufacture improved and made easier, thus increasing the rate of production, and the amount of material lessened; whilst the strength may be well maintained. On the other hand, carelessness in detail design may quite conceivably nullify any good points in the general scheme, it may increase the weight of machine, and may also weaken an otherwise strong design.

For instance, a wing joint, or a fuselage fitting, if incorrectly designed, may give persistent trouble in the aeroplane, due to distortion or deflection when in flight or on landing.

There are a few general maxims which might be given as a preliminary to this chapter, and which are capable of quite general application. They are—

(1) Make sure of the strength of the materials used.

(2) Avoid unnecessary changes of section, no matter what may be the material, and avoid sharp bending of thin metal plates.

(3) Do not drill unnecessary holes in vital parts, and keep all holes as small as practicable.

(4) Let the strength of a fitting be exactly where the maximum load is applied.

(5) All wiring plates, lugs, strut fittings, etc., should be at the correct angle of the respective wire or strut so that deformation may not occur.

(6) Arrange, as far as can be done, that offset loads may not exist.

(7) Reduce the weight but maintain the strength.

(8) Simplify the design to the uttermost.

It may be said that these are easier to quote than to apply, but the nearer they can be approached the better structure will be evolved; they may be considered as ideals at which to be aimed rather than easily attained. In order to give proper consideration to the points as they arise, it is proposed to take various components in some sort of order.

MAIN PLANES

Spars

Spars are now made in various forms, although originally they were almost invariably cut from long solid pieces of timber suitably spindled at points between spar fittings into the common *I* section, being left solid where bolt holes are to be drilled through the spar. This form still exists, quite frequently, but recently, owing to the serious shortage of spruce, spars have been built up of several lengths of specially cut

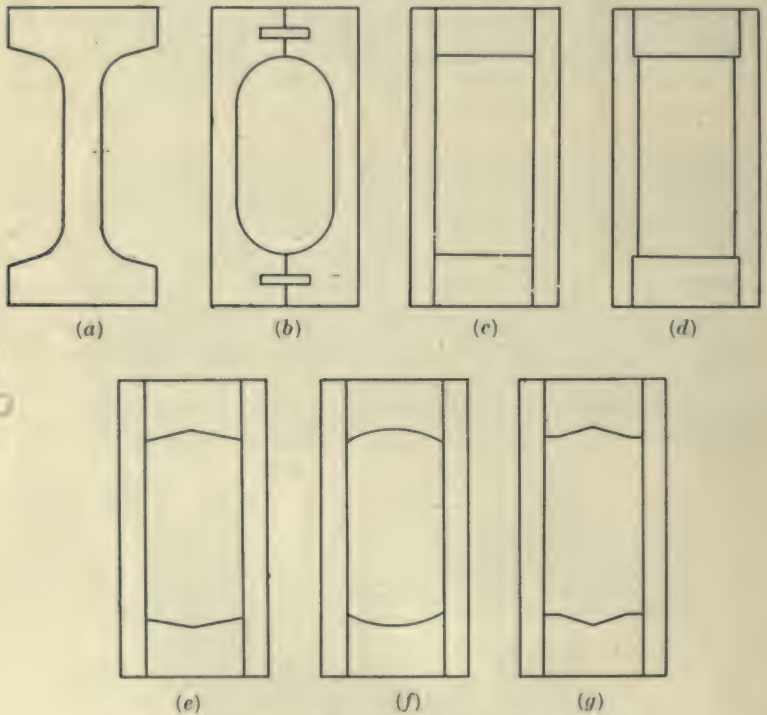


FIG. 93.—VARIOUS SECTIONS OF BUILT-UP SPARS

timber, glued together, and, in some cases, carefully wrapped with glued fabric.

This latter type is now very popular, and is slightly varied in its form. Fig. 93 shows several sections, of which (*c*) is the best, since it is comparatively easy to construct, and is also the strongest of the built-up sections ; (*d*) is more difficult to make so as to be certain that the joints are good, and is less reliable. Types (*e*) and (*f*) are variations of (*c*) with a view to obtaining greater glued surface where the webs are attached to the flanges. This advantage is, however, more imaginery than real, and the spar is more costly to produce than (*c*), hence it does not warrant the extra trouble.

In (*e*) and (*f*) the sharp corners of the flanges are apt, on account of the way in which the timber is cut, to leave the webs and probably decrease the glued surface rather than increase it, and so pave the way to further weakness.

Type (*g*) is an attempt to overcome the objection just quoted, but is hardly successful enough to warrant the change from (*c*). Type (*b*) is good, but not as simple as (*c*) ; (*a*) is the ordinary *I* section spindled out from the solid rectangular section.

It may be pointed out that type (*c*) has a larger lateral moment of inertia than (*a*), and this is valuable in the consideration of lateral failure of spars.

Often the webs are screwed or carefully tacked to the flanges, but not much advantage is to be obtained by so doing ; the better scheme is probably to wrap carefully with glued fabric.

If long lengths of timber are unobtainable, joints may be made in either the flanges or webs by splicing. The angle at which the splice is made should be small, in order that a good surface may be provided for glueing.

Care should be taken that no two splicings occur in the same vertical section of the spar.

Solid places in the spar should be made as shown in Fig. 94 in order that no sudden change of section may occur. The length of such a solid piece depends upon the length of the metal fittings, and also upon the magnitude of the bending-moment and the shearing force at that point on the spar.

Ribs

Ribs must be so designed that the reaction due to air lift forces may be safely transferred to the spars. Fig. 95 shows the popular type of light rib in use at present, in which the once common ply-wood web is abandoned. In this example, the upper and lower booms are connected at intervals by wooden posts slit top and bottom to receive the narrow portions of web to which they are carefully attached by glue. This is quite a satisfactory type of section for small single or two-seater machines where the maximum applied load, including factor of safety, would not exceed about 75 lbs. in shear. It will be observed that the web portion is sunk into the boom (Fig. 96). This web portion is cut away where the spar passes through the rib, for the spar must not be cut down on any account.

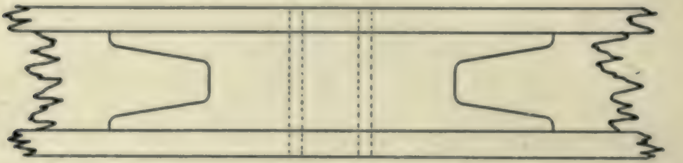


FIG. 94.—SOLID PLACE IN SPAR

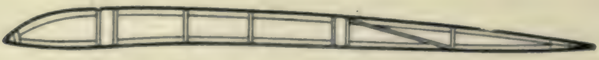


FIG. 95.—POPULAR TYPE OF RIB

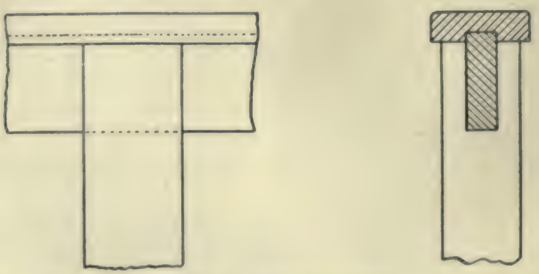


FIG. 96.—DETAIL OF RIB CONSTRUCTION. ATTACHMENT OF WEB TO BOOM

Stronger vertical posts are, however, inserted on either side of the spar and fixed thereto, as well as to the rib flanges by screws and glue.

As the size of an aeroplane increases, the required width of rib flange does not suffer much increase, because the available depth of the wing section is greater, which in turn allows of a deeper, and consequently stronger rib.

The load, on the other hand, is probably several times greater than that already quoted for a small or medium sized machine, consequently, special attention must be paid to the manner in which the connection is made between the rib and the spar in the larger machine.

There are several ways in which satisfaction may be obtained. The rib may be first fixed to the spar in the usual way and then carefully bound with glued tape, so that there would be no tendency for the flange to leave the web. If the solid *I* section spars are used (Fig. 97), then the vertical posts in front of and behind the spar may be extended so as to fit into the spindled part of the spar, but this is expensive, since it means increased work by hand.

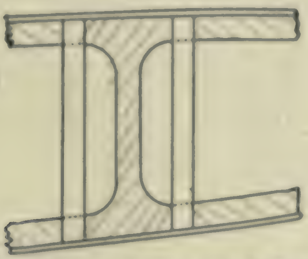


FIG. 97.—ATTACHMENT OF RIB TO *I* SECTION SPAR

If the spars are built up on the Tarrant principle (*i.e.*, the girder type), then a piece of ply-wood could be made to just pass between the flanges of the spar (*i.e.*, through the braced web portion of the girder spar) connecting the vertical posts of the rib. The spacing of ribs along the wing is important, and never ought to exceed 18 ins., an average being somewhere between 12 ins. and 18 ins., according to size of wing. On seaplanes, however, the rib spacing on the bottom wing must be reduced somewhat, in order to withstand the heavy loads sometimes induced by sea water. Too wide spacing means excessive sagging of fabric, with consequent loss of efficiency.

Nose ribs are always inserted in main planes, and the spacing of such nose ribs is one-half or one-third that of the full ribs. Their function is to support the fabric at the point of entry of the plane into the wind.

Drag Bracing

As will be seen from the G.A. views inserted between pages 8 and 9, the front and rear spars are to be braced so that drag loads may be taken. On this account, certain of the ribs need strengthening to be capable of acting as compressive members (*i.e.*, drag struts). An alternative to the strengthening of the ribs is to introduce special compressive members in the shape of steel tubular or wooden struts.

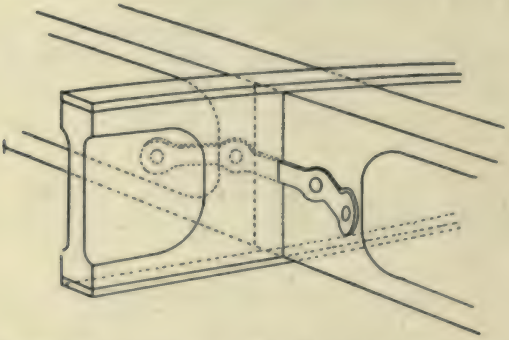


FIG. 98.—COMPRESSION RIB CONNECTION TO SPAR

The bracing is generally composed of round steel wires with swaged ends, attached to wiring lugs against which the compression members, in whatever form, ought to abut. Fig. 98 illustrates a typical arrangement at such a connection between the wire, spar, and compression member, when the compression member is a strengthened rib. Fig. 99 shows a similar connection when a special compression tube is introduced. It will be seen that both schemes necessitate one or more bolt holes laterally through the spar, and it is advisable that these holes be set as near as possible at the middle of the depth (*i.e.*, at the neutral axis, for this is quite important).

Connection of Struts, etc., to Spars—Spar Joints

Fig. 100 shows a common method of attaching an interplane strut to the spar by means of an eyebolt.

In cases (*a*) and (*b*) the major axis of the spar is normal to the wing chord. If the wings are not staggered, an ordinary eyebolt will be sufficient, as in (*a*), but if the wings are staggered the spar is subjected to torsion of amount Px, since the line of strut thrust does not pass through its neutral axis. Further, the ordinary type eyebolt, if used under such conditions, will be subjected to a bending-moment Px^1 at its root, while the bearing pressure of the bolt in the spar (due to the horizontal

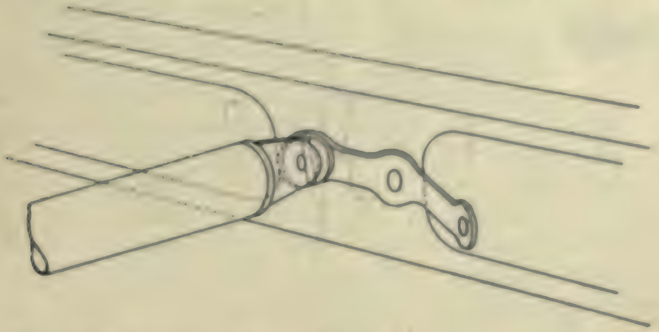


FIG. 99.—TUBULAR DRAG STRUT CONNECTION TO SPAR

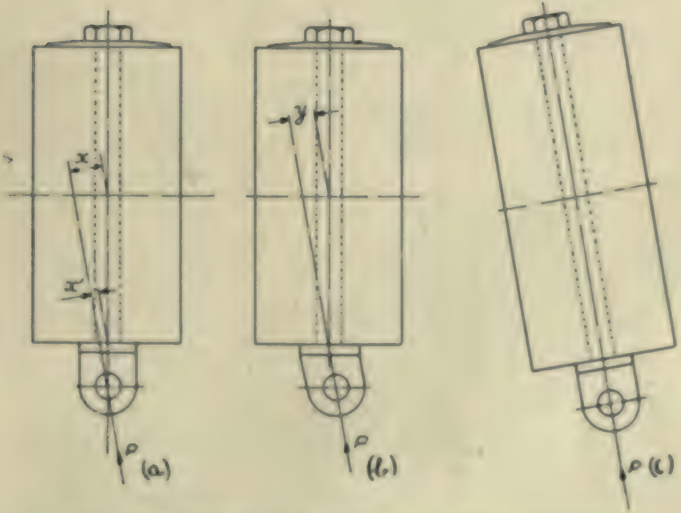


FIG. 100.—STRUT ATTACHMENT TO SPAR EYEBOLT

component of *P*) should be checked. In cases where it is excessive, ferrules are often used so as to increase the bearing area. In large machines, hollow bolts of larger outside diameter are sometimes used, but these are comparatively expensive to produce.

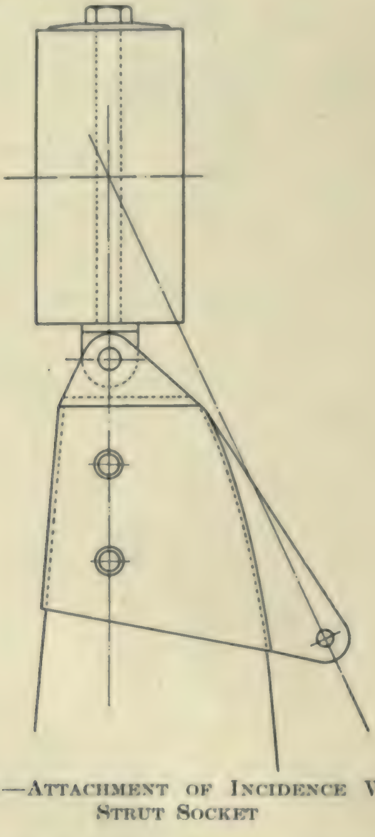


Fig. 101.—Attachment of Incidence Wire to Strut Socket

A better arrangement is shown in (*b*), where the eyebolt is specially made so that there is no offset bending-moment at its root. The spar will still be in torsion, but this can be taken up by the ribs.

Sometimes the spars are arranged as in (*c*), where there is no torsion in the spar, and no bending-moment to be taken by the eyebolt.

The method of attaching the main lift wire has already been indicated in Fig. 9. Here, it is important to see that the bearing pressure of the bolts in the spar due to the horizontal component of the lift wire load is within reasonable limits.

The wiring plate, being underneath the eyebolt carrying the interplane strut, serves to reduce the bearing pressure on the spar due to the strut thrust and due to tightening up.

If the interplane strut were attached to the other bolt, the strut load would be transferred directly to the lift wire, and the question of bearing pressure would not apply, except in so far as there would be a local lift at that point. The effect of such an alteration would be to increase the offset bending-moment on the spar, in which case it is necessary to see that the spar can take it. The point of support in this case would be at the intersection of the line of thrust and the neutral axis of the spar as before. The incidence wires are usually attached to lugs provided on the strut socket, as indicated in Fig. 101.

Sheet Metal Type

This type, consisting of one or more wrapping plates, may be used with advantage on all sizes of machines, but particularly on large ones.

For easy manufacture of the plates, the spar can be locally packed square. The main lift and incidence lugs are all bent out of the plates, the lugs for the interplane strut being slipped through slots in one of the plates. The arrangement is such that the plates can be fitted or removed after the ribs are in position. The clip as a whole is held in position by bolts, the number varying according to that required.

It is shown in Fig. 102. This form has some disadvantage, inasmuch as it often means that holes running laterally through the spar are required at some distance from the neutral axis, but no harm is done so long as the actual spar flanges are not drilled. The lugs from such a fitting, which take drag struts, must of necessity be made much stiffer and stouter than in the case of tension wiring lugs, so as to take compressive load without failure; such lugs are stronger if kept as short as possible. The bearing area of the strut pin in the fitting ought also to be made quite sufficient.

In the case of all fittings attached to spars by bolts passing through the spars, the friction is worth while taking into account, as helping the bearing of the bolt on the wood. The co-efficient of friction is generally taken at about ·3, and the area over which friction may be assumed to be operative is that portion which the bolt may reasonably be assumed to control. The bearing pressure of the fitting on the spruce spar due to tightening up ought not to exceed 800 lbs. per square inch, and this figure should be taken as the normal load in estimating the frictional load.

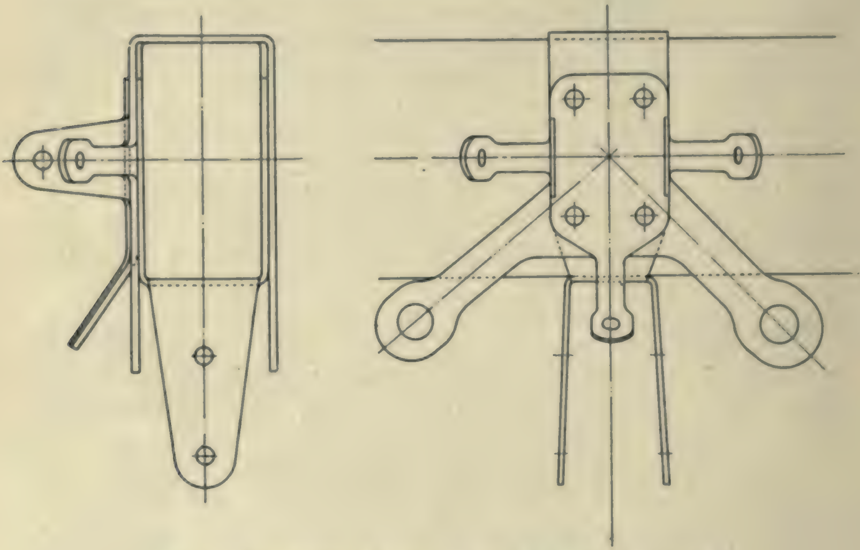


FIG. 102.—SHEET METAL TYPE OF SPAR FITTING

Thus, if A = area controlled by bolt in square inches, then allowable frictional load, which may be taken as helping the bolt in bearing on the spar, is—

$$A \times \cdot 3 \times 800 = 240\,A \text{ lbs.}$$

Very frequently this amounts to more than the load taken directly by the bolt in bearing.

For simplicity the designer ought to endeavour to reduce the number of plates in such a joint to a minimum by discreet bending and shaping, etc.

Tension Lugs

A typical form of tension lug is shown in Fig. 103.

The ideal in the design of such lugs is to obtain uniform strength throughout. Thus, the lugs must be as strong in bearing of the pin on the plate, in tearing out the end of the lug, in tension across the hole (which latter must be large enough to satisfy the pin in shear), as it is in tension at its parallel portion. These are usually worked out and standardized once for all by the designer, since they are in such constant use, and when listed each is generally known by the size of wire for which the particular lug is suitable.

It is, perhaps, advisable to put limits on the various dimensions of such lugs, especially the pin hole. Scales of limits, however, are almost as numerous as designers.

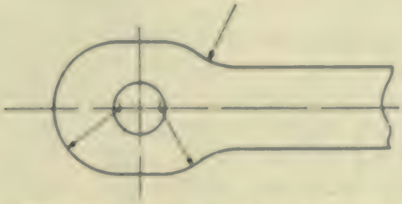


FIG. 103.—TYPICAL FORM OF WIRING LUG

The usual clearance for the hole of wiring lugs is about $\frac{1}{64}$ in., but even this varies. Where extra bearing surface is required, a thickening piece is brazed to the end of the lug, as in the case where one wiring plate has two lugs taking differently sized wires.

Sometimes this thickening piece is carried along the whole lug and underneath the bolt head, so that not only is the lug helped locally, but the whole lug is strengthened.

Fig. 104 shows the two methods of strengthening wiring lugs.

Bending of Wiring Plates, Etc.

It is very important that lugs be bent at the correct angle of the wire or strut to which it is to be attached, and, furthermore, it is important that the bending line be not far away from the head of the bolt by which the plate is fixed to the spar.

The reason for this is very obvious from Fig. 105 (*a*) and (*b*), which show the correct and incorrect methods of bending such a lug. Fig. 105 (*c*) shows lug set at incorrect angle.

It is easily seen that unless the lug is pulling right from the head of the bolt there is the tendency for the lug to " give " when the wire is subjected to maximum load, in which case the

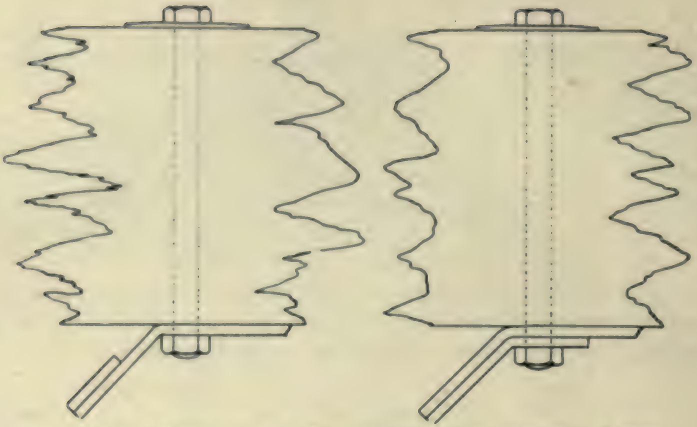


FIG. 104.—METHODS OF STRENGTHENING WIRING LUGS

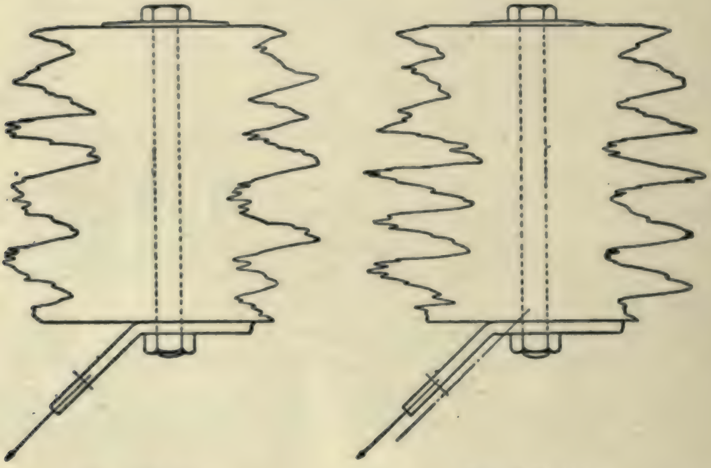


FIG. 105 (*a*) FIG. 105 (*b*)

(*a*) CORRECT. (*b*) INCORRECT POSITION OF BEND IN WIRING PLATE

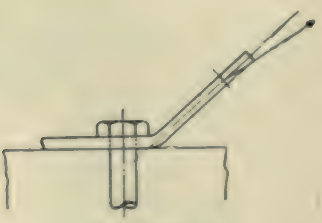


FIG. 105 (*c*)

wire will slack off and the structure will, consequently, be deformed. The importance of making the line of pull pass through the point of support (*i.e.*, at the junction of neutral axis of spar and strut line) has been impressed in another place.

Bending allowances ought always to be made so that easy bends may be attained. Sharp cornered bends should always be avoided. Even in the sharpest bend, the minimum radius should not be less than ·05 in., or the thickness of the plate.

Holes in Spars

Wherever eyebolts are used as strut attachments, it means that the bolt holes required are necessarily in the direction of the major axis, and, furthermore, as is sometimes the case, when ferrules are used, these holes are slightly larger than the diameter of the bolts themselves. The direct effect of the boring of holes through a spar in this direction is, of course, to reduce seriously the moment of resistance of the spar to bending, and that is the reason why the stress at such a point on the spar is always estimated in order that it may not exceed the limiting value. To some extent any additional wood packing pieces glued to the spar, or metal fitting plates attached at such a point may contribute to the strength there, but these are usually neglected in calculations. Spars are always made solid at such points and for some short distance on either side, because the shearing stress is often fairly high in the neighbourhood of a support. The usual alternative to the use of eyebolts is the application of wrapping plates. These are bent into the form of U and bolted laterally through the spar, lugs being rivetted and brazed to these plates to receive struts. If the bolt holes are drilled through the spar laterally at or near to the neutral line, then the moment of resistance to bending is not much reduced, and, on this consideration alone, they are not of much importance, but when it is considered by how much the local stress may possibly be increased by the drilling of such a hole, even close to the neutral axis, it is essential that the number and size of such holes should be kept as small as possible. It was Professor Coker who first pointed out (*see Engineering*, March, 1912) by his optical stress methods that if in a beam a central hole be drilled laterally, then the stress at the outer edge of the hole would be greater than that on the assumption of its being proportional to the distance from the neutral axis. It was stated that if the diameter of the hole be one-third the depth of the spar, then the stress at the outer edge of the hole, instead of being one third the skin stress, is actually 52% of that stress (*i.e.*, about 57% greater than if it obeyed the linear law, *see* Fig. 106).

In the case where two holes are drilled, the variation of stress is still more complex, and at the outer edge of the hole is actually about 55% more than the skin stress at the outer edge of the beam. At the inner edge of the hole, the stress is greater than that given by the linear law.

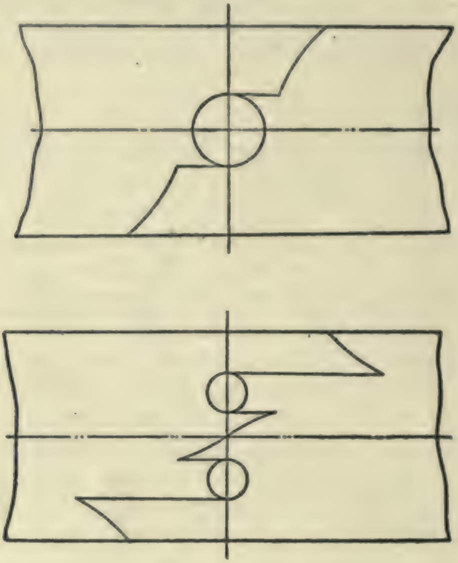


FIG. 106.—SHOWING VARIATION OF STRESS DUE TO HOLES IN SPAR

These examples show the importance of keeping all such holes as few and as small as possible.

The ways in which a joint may fail are—

(1) Shear of the bolt.
(2) Tension of the bolt.
(3) Shearing through the timber.
(4) Crushing of the timber under the bolt.

Items (1) and (2) obviously depend upon the size of bolt, but it is possible sometimes that the bolt may be strong enough in tension and shear, and yet not present sufficient surface to the timber to avoid failure under items (3) and (4). To overcome this, as has been seen, ferrules are often used, or, in large machines, hollow bolts of larger diameter could be adopted.

The following are experimental values* which agree with those calculated for mild steel bolts in good quality spruce, where the bearing length is not less than five times the bolt diameter—

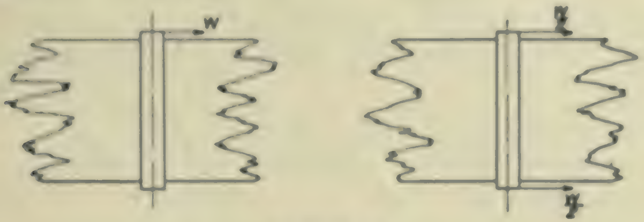


TABLE 26

Load acting.	On one end of bolt.		On both ends of bolt.		
Diameter of bolt. Inches.	*W* lbs. Bearing on side grain.	*W* lbs. Bearing on end grain.	Diameter of bolt. Inches.	*W* lbs. Bearing on side grain.	*W* lbs. Bearing on end grain.
1/8	60	150	1/8	125	250
3/16	120	300	3/16	250	575
1/4	220	540	1/4	425	1050
5/16	350	850	5/16	700	1650
3/8	480	1220	3/8	950	2350
7/16	650	1660	7/16	1300	3200
1/2	850	2160	1/2	1700	4200
9/16	1075	2740	9/16	2150	5300
5/8	1350	3400	5/8	2700	6530
11/16	1600	4150	11/16	3250	7800
3/4	1850	4850	3/4	3700	9200
13/16	2100	5600	13/16	4200	10600
7/8	2400	6400	7/8	4800	12000
15/16	2700	7200	15/16	5400	13600
1 in.	3000	8000	1 in.	6000	15000

Use of Pins in Fork Joints

Parallel pins are used very considerably in aeroplane work, and it is important to see that they are well able to support the forces to which they are subjected. Where a pin is well fitting, and can be maintained at that degree of excellence, shearing is all important, provided the bearing area has been made satisfactory. In such a case of double shear of a well-fitting pin,

---

* The results of experiments carried out at the R.A.E. were plotted and a fair curve drawn, from which curve the values in the above table were obtained.

$$P = \frac{C\,\pi d^2 f}{4}$$

where P = applied load, including factor.
d = diameter of pin.
f = allowable shear stress.
C = constant 1·75 to 2 for double shear.

Unfortunately, however, the fitting of pins in holes becomes loose after a time, due to wear, and then it is necessary to take into account the inevitable bending which undoubtedly occurs. The general method of treatment given by Unwin* and other technical writers is as follows—

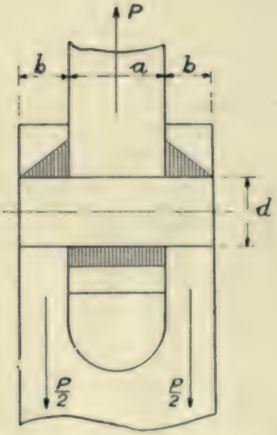


Fig. 107.—Pins in Fork Joints

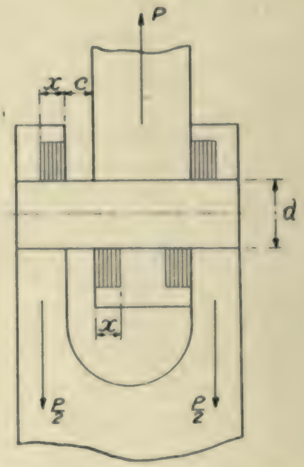


Fig. 108.—Pins in Fork Joints

The load on the length a of the pin is assumed uniform and that on the fork b is assumed triangular (*see* Fig. 107), *i.e.*, intensity of load increasing uniformly from outer edge inwards.

Then the bending-moment at the centre of the pin will be—

$$\text{B.M.} = \frac{P}{2}\left(\frac{a}{4} + \frac{b}{3}\right)$$

---

* *See* Unwin Machine Design, Part I.

where P is the load on the pin, including load factor on machine. The section modulus of the pin

$$= \frac{\pi}{32} d^3$$

hence, stress due to bending will be

$$= \frac{\frac{P}{2}\left(\frac{a}{4} + \frac{b}{3}\right)}{\frac{\pi d^3}{32}}$$

which must be kept within the allowable stress for the material.

Another treatment of loosely fitting pins which the Authors suggest is the following—

It is assumed here that for a distance x, as shown in Fig. 108, the material of the fork and tongue is stressed up to its maximum allowable intensity of bearing pressure.

Then let—

$P =$ load on joint in lbs.
$d =$ diameter of pin in inches.
$f_B =$ allowable bearing pressure on material, lbs./ins.^{2}
$M =$ allowable bending-moment taken by pin in lbs., ins.
$c =$ clearance, which is generally small compared to the distance between the jaws of the fork.

From bearing considerations, we have

$$2\ df_B x = P, \text{ or } x = \frac{P}{2df_B}$$

but

$$M = \frac{P}{2}(x + c)$$

approximately for the bending of the pin, and inserting for x, we have

$$M = \frac{P}{2}\left(\frac{P}{2df_B} + c\right)$$

$$= \frac{P^2}{4df_B} + \frac{P_c}{2}$$

or

$$P = \sqrt{4df_B M + d^2 f_B^2 c^2} - f_B dc$$

If c is so small as to be negligible,

$$P = 2\sqrt{df_B M}$$

Now the maximum bending stress taken by the pin depends upon the modulus of resistance of the section.

For a circular section,

$$Z = \frac{\pi}{32} d^3$$

then if f = maximum allowable stress in the pin due to bending, we have $M = f\,Z$.

Hence, this value may be inserted in the formula just deduced and the following table compiled.

TABLE 27

Values of P for different clearances for bolts of various diameters are tabulated below for p bearing pressure and bending stress of 80,000 lbs. per square inch.

Diameter.	0	0·1 in.	0·2 in.	0·3 in.
$\frac{1}{8}$ in.	775	265	145	100
4BA (0·142) ,,	1000	370	215	145
2BA (0·185) ,,	1740	805	470	330
$\frac{1}{4}$,,	3140	1720	1085	770
$\frac{5}{16}$,,	4900	3025	2000	1400
$\frac{3}{8}$,,	7000	5000	3400	2400
$\frac{7}{16}$,,	9500	7000	5000	3800
$\frac{1}{2}$,,	12530	9160	6870	5350
$\frac{9}{16}$,,	15700	12400	9300	7200
$\frac{5}{8}$,,	19400	15500	12000	9600
$\frac{11}{16}$,,	23700	19000	15100	12400
$\frac{3}{4}$,,	28250	22850	18700	15500
$\frac{13}{16}$,,	33500	27300	23000	19000
$\frac{7}{8}$,,	39000	32500	27300	23000
$\frac{15}{16}$,,	44500	37500	32000	27300
1 ,,	50150	42800	36700	31600
$1\frac{1}{16}$,,	57000	49000	42500	37000
$1\frac{1}{8}$,,	64000	55500	48500	42500
$1\frac{3}{16}$,,	71000	62000	54500	48000
$1\frac{1}{4}$,,	78500	69000	61000	54000
$1\frac{5}{16}$,,	87000	77000	68500	61000
$1\frac{3}{8}$,,	95000	85000	76000	68000
$1\frac{7}{16}$,,	104000	93000	84000	75000
$1\frac{1}{2}$,,	112800	101500	91500	82300
$1\frac{9}{16}$,,	122500	111000	100000	90500
$1\frac{5}{8}$,,	132500	120500	110000	99500
$1\frac{11}{16}$,,	143000	130500	119000	108500
$1\frac{3}{4}$,,	153500	140500	128200	117500
$1\frac{13}{16}$,,	165000	151500	138500	127500
$1\frac{7}{8}$,,	177000	162500	149500	137500
$1\frac{15}{16}$,,	188500	173500	160000	148000
2 ,,	200500	185000	171000	158000

Attachment Fittings

At the four points where the spars are attached to the fuselage, the attachment fitting must be examined carefully, because end load is often very considerable there, either as tension or compression. The bottom plane is usually hinged to the fuselage, the hinge pin being set horizontally, so that no bending-moment is taken at that point. The hinge pins ought, however, to be carefully checked for bearing, shearing, and bending of the pin due to loose fitting. The tearing of the fitting plates should also be checked. On the upper plane at least one of the wing attachment fittings is made fixed, the rear joint may or may not be hinged as on the bottom plane. At fixed joints, top and bottom fish plates are secured to the spar by vertical bolts, the outer and centre portions of the spar butting together closely.

This attachment fitting must withstand whatever bending-moment there is in the spar at that point.

Miscellaneous Details on Main Planes

The leading edge is usually made up of ash and is suitably shaped to receive the flanges of the ribs, which are glued and screwed, or otherwise fixed by means of metal clips. A common section of leading edge is shown in Fig. 109; often it is hollowed in the lengths between the ribs.

The trailing edge is generally made of spruce and attached to the ribs as shown in the same figure. As an additional support to ordinary ribs, a tape is set zig-zag under and over the ribs, right along the length of the plane and back again, midway between the spars (*i.e.*, the tape passes under one rib and over the next, and so on).

At the points where the tape is in contact with the flange of a rib, it is glued and tacked to it. To some extent, this scheme enables one rib to support another laterally. The main idea is to connect all light ribs to strength or end ribs, so that all the ribs forming the wing would deflect together laterally.

To assure that the light ribs are receiving lateral support (if necessary) from strength ribs, all ribs in the wing are connected by means of stringers. These are usually of spruce, about ¼ in. square in section; two would be used on each wing, being placed between the spars. One stringer would connect all the upper booms, and one the lower booms of each rib. Fig. 110 shows the stringers in position and the chocks glued on to them on each side of the web. Sometimes a stringer is placed between the rear spar and trailing edge.

The inner end ribs of ailerons are usually made considerably stronger than ordinary ribs, and very often a diagonal stay piece is set across the corners of this inner end bay.

At the outer end of the aileron, the edge is generally curved, and is frequently made of an ash bend, but ash bends are tolerably expensive to make, and warping is so difficult to avoid, that spruce pieces have been introduced. Fig. 111 shows how this has been done.

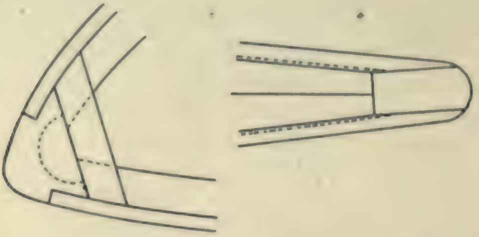


FIG. 109.—LEADING AND TRAILING EDGES

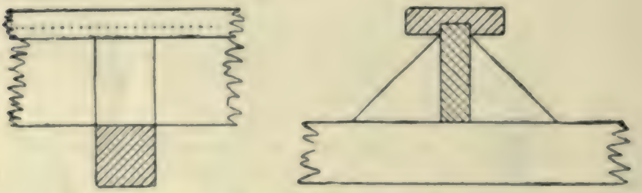


FIG. 110.—ATTACHMENT OF STRINGERS

It is important that wherever possible all nuts on bolts used on main planes for fixing of joint plates, wiring lugs, strut lugs, etc., etc., should be locked by carefully rivetting over. Washers should be used in places where the bearing stress due to the initial force from tightening up, or the actual load coming on the bolt would be otherwise excessive. This is particularly so in the case of bolts through spars, where a larger aluminium or duralumin washer is often inserted to protect the spar against crushing of the wood. Plain pins and fitting bolts are generally locked by means of split pins, and taper pins must always be rivetted over.

Bracing in the main planes is not always made of swaged wire with adjustable end fittings, but is occasionally made of plain piano wire of suitable strength, adjustment being made by means of turn-buckles set at one end of each length of wire.

Where a turn-buckle is attached to a wiring plate, a fork end taking the plate would be used, but where the turnbuckle connects up to a wire, an eye would be used. Turnbuckles are made in standard sizes to suit standard strengths of wire.

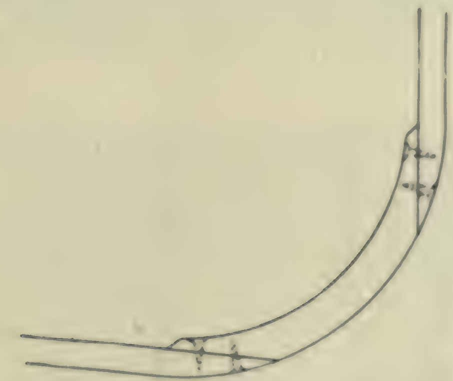


FIG. 111.—BENDS FOR AILERONS AND ELEVATORS

Similarly, when a plain piano wire is attached directing to a wiring plate, the hole in the latter must be comfortably rounded out to avoid abrasion of the wire by sharp corners.

Struts in wing structures are occasionally made of metal tube, either streamline, as in Austin "Whippet," or of round tubes with wood fairing as in some German types. The relative weight must be carefully considered, however, or the structure may be unduly heavy. In the "Whippet" all flying and incidence wires were abolished and replaced by streamline tubular struts. Regarding metal construction, the reader is referred to valuable work published in *Aeronautical Journal*, in September, 1919.*

* "Metal Construction of Aircraft," A. P. Thurston, D.Sc., *Aeronautical Journal*, Sept. 1919.

FUSELAGE

The fuselage may be one of three types, viz.—

(*a*) Girder type, consisting of longerons, horizontal and vertical cross members all braced together by wires to form a rigid structure. The material may be either wood or metal tube.

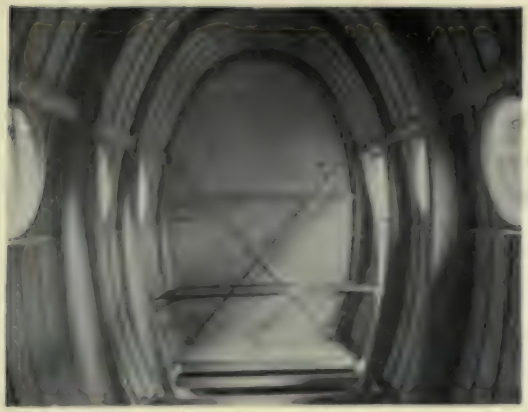
(*b*) Semi-monocoque type, in which a number of longerons are held in position by hoops or rings suitably spaced along the length of the structure. The whole is then covered by thin ply-wood, which acts as a covering possessing stiffening properties.

(*c*) Pure monocoque type in which the fuselage is merely a wooden shell.

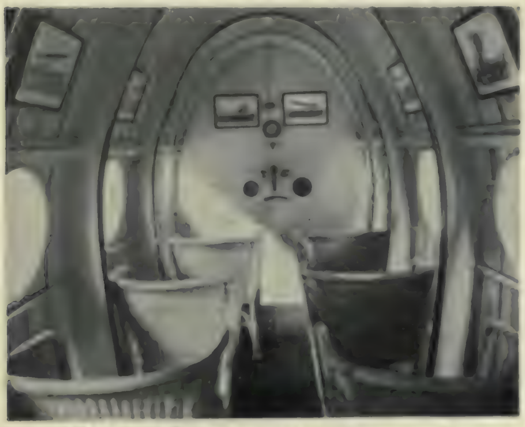
The metal tube type is at present confined to tubular members without wire bracing, and thus really forms a special case of the girder type.

We shall now briefly examine each of these types and note their advantages.

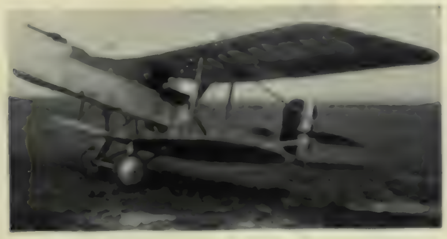
The most common is the braced type (*a*) in which the longerons and struts are of ash or spruce with swaged wire adjustable bracing in the various bays. A joint of such a fuselage is analogous to Fig. 102, where provision would be made for the attachment of side, top, and bottom, and diagonal bracing. A few of the bays on the top of the fuselage which cannot be directly braced by means of wires are stiffened in other ways; this applies also to some of the diagonal bays. The metal fittings invariably require one or more holes drilled through the struts and longerons, and in this connection the designer ought to bear in mind what has been said regarding holes in spars. If the load in a strut is fairly large then the bearing pressure between this strut and a longeron must be improved by inserting a piece of metal plate between the two, or by arranging the metal fitting so that this bearing pressure is of an allowable intensity; this is important if the strut be of ash and the longeron of spruce. This type of fuselage is easy to manufacture and erect, and its square or rectangular section is advantageous as regards the attachment of planes, undercarriage, tail plane, etc. In many cases the front portion of the structure is made of metal tube to support the power unit, but there is very little to recommend such a



VICKERS' "VIMY" (COMMERCIAL)—VIEW OF INTERIOR, SHOWING CONSTRUCTION



VICKERS' "VIMY" (COMMERCIAL)—VIEW OF INTERIOR OF CABIN, FRONT END



F.E.9 AEROPLANE WITH 150 H.P. HISPANO ENGINE

practice unless it be that bracing wires can be dispensed with, consequently there will be little or no deformation in that portion.

The advantage of the braced wooden type of fuselage is that it can be built with a minimum of weight, since all the wood parts may be spindled. It forms a good strong structure and possesses capability to withstand shocks.

The bracing wires are, however, somewhat of a disadvantage on the score of—

(1) Cost of manufacture.

(2) Liability to stretch, thus producing deformation of the structure which may interfere with the setting of flying or stabilizing surfaces.

Semi-Monocoque Type

This type is practically a combination of the girder and monocoque type, whereby no bracing wires are required. The construction consists of a number of longitudinals of built up or *I* section, held in position by hoops at various intervals, the whole being covered by ply-wood, glued and screwed to the longerons and hoops. The circular section fuselage lends itself more particularly to this type, but it would be quite possible to have an oval or even rectangular section.

In the design the longerons are made strong enough to take any bending due to landing loads or forces on the tail, while the ply-wood is assumed to take the shear. The function of the rings is to enable the structure to maintain its shape, and also to hold the longerons in position and to keep them well up to their work. Hence, the spacing of such rings will be largely determined by the amount of support necessary to be given to the longerons on the compression side of the fuselage so that they will not fail as struts at the points of attachment of wings, tail plane, undercarriage, etc., where extra local strength is required. This is obtained by the use of larger hoops or two ordinary ones at such places. The ply-wood covering may be cut away more or less as required, for the admission of pilot, etc., but it is important that not too much be taken from the sides of the structure where the maximum shear stress occurs. Plate 8 gives a good idea of the type of structure, and it will be observed how much more room there is in a fuselage of this kind as compared with the girder type where diagonal bracing wires are necessary.

Flying boat hulls are constructed on similar lines, but they must possess not only strength but a certain amount of elasticity, in order to withstand heavy loads due to sea water forces.

Pure Monocoque Type

This type is not very popular, although it made its first appearance many years ago.

As the name indicates, this type consists merely of a wooden shell. There is an entire absence of bracing wires and longerons, so that the shell, usually of ply-wood, must be of such thickness as to withstand the loads the fuselage is called upon to bear.

The great advantage of this type is that it can be made to any desired aerodynamic shape, whereas in other types the shape is more or less defined (*i.e.*, square, rectangular), unless fairing is secured to the fuselage structure. There are other advantages in common with the semi-monocoque type already mentioned.

Metal Tube Type

This type, in the relatively few examples in which it has been tried, has been very satisfactory.

The fuselage may be made in one continuous length or may be constructed in two sections, front and rear, which may be made easily detachable if necessary. It is made square or rectangular in section, and the longerons are continuous lengths of tubing upon which fittings are slid and set according to the distances between the various cross members. The cross members also are of tubular form, reinforced at the ends by a short piece of telescopic tubing.

The ends of these cross members are flattened, and on erection fit inside the lugs provided on the fittings already mentioned. No bracing wires need be used, as will be seen in the example on Plate 5, which shows quite well how the sides and top and bottom of the fuselage are braced. This photograph is actually a picture of the Austin " Whippet " in skeleton. At the connections of the cross tubes to the fittings, small bolts or solid or hollow tubular rivets may be used.

Plate 5 shows a typical fitting for such a fuselage, which fitting may be standardized for several joints if carefully arranged.

If and whenever this can be done, considerable economy will be effected. The fittings may be sweated and pinned to the longerons.

This type of fuselage has proved itself to be quite satisfactory from a strength point of view, and is not liable to distortion in the same way as structures braced by means of wires, since there is less possibility of the members elongating under load. The weight of the structure will be somewhat greater than for the wire-braced structure with wooden members, but the difference ought not, strength for strength, to exceed about 10%. One or two difficulties present themselves on structures of this type at joints where other components are attached, namely, landing gear and spar attachment fittings on the fuselage, and there is room for considerable ingenuity in design at such points. The reinforcement of the ends of cross tubes demands attention, because such members in compression would be weak at the point where flattening takes place, the gauge of the tube material being comparatively small.

The method of reinforcing such tubes is effected by telescoping a short length of tube on the end of the member, either externally or internally, before flattening and subsequent dip brazing. The short reinforcing tube is generally placed inside the main tube and then flattened in the press, dip-brazed in a brazing bath, and afterwards drilled to jig length. Since in this method only the ends, to a distance of, say, 2 to 3 ins., are dipped into the bath, when brazing it is a good plan to carefully anneal the tube afterwards. A small hole $\frac{1}{32}$ in. diameter ought to be drilled into the tube (but not through both sides) as far from the centre of its length as is practicable, in order to allow of the free escape of air from, and the consequent flow of brazing metal into, the reinforcement.

Fig. 112 will show how the reinforcing tube is inserted and its appearance after flattening. This type of fuselage can be very quickly assembled to jig and may be considered far more of a permanent job than the ordinary braced type. No adjustment is possible, however, and in general is never required.

The main points of advantage of this metal construction are strength, durability, and ease of construction.

As an alternative to the use of tubes it is not improbable that fuselages may be constructed of several light longerons made from sheet metal bent and pressed into suitable sectional outline, and held together by bulkheads also constructed from sheet metal.

In each of these types of construction, the method of mounting the engine calls for some deliberation, depending upon the type of engine, whether of the Vee type or the radial type.

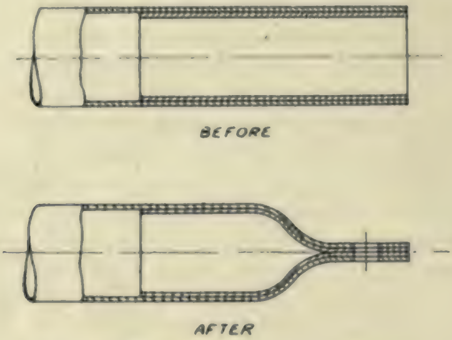


Fig. 112.—Tube Reinforcement

The Vee type of engine is accommodated on two stout horizontal engine bearers, usually of ash, to which it is held by bolts through lugs provided on the crank case. These engine bearers are in turn supported by bulkheads connecting the two sides of the fuselage, which are stiffened by means of fairly strong three-ply ash on either side, and metal plate fittings complete the arrangement. Such a scheme is practically universal and is entirely satisfactory. Sometimes, however, the engine is designed to take tubular bearers, in which case the engine mounting would be entirely of steel tube. In the case of the radial engine a different type of engine attachment must be sought. Usually a fairly stout flat plate, flanged all the way round the edges with a suitably flanged hole in the centre and other lightening holes also flanged in a lesser degree, is designed so that it will fit over the front end of the fuselage to which it is then bolted. The flanging of the plate stiffens it to some extent and the engine is directly bolted to the plate by a ring of bolts set round the central hole, the bolts passing through solid lugs on the engine crank case. This arrangement is good so far as merely supporting the engine is concerned, but the thrust of the propeller must be taken in some other way than by a flat plate. This may be done quite well by means of about four or six tubes from the ends of the longerons of the fuselage to points on the crank case of the engine ; they are thus able to transmit the propeller thrust to the fuselage direct, and are further able to help the plate in the supporting of the engine with its centre of gravity overhung from the fuselage. It is an advantage if these stay tubes have means of adjustment so that they may be lengthened or shortened for convenience.

In the different types of fuselage these engine mountings will of necessity be correspondingly modified. The static thrust of the propeller may be taken to be about 7 lbs. per H.P. developed.

Tubes in Tension

With tubes in tension, some amount of care is necessary in the adoption of the type of end fittings. The following are standard methods.

If it is not desirable to flatten the tube, then a socket may be turned to a good fit into the ends of the tube and pinned and sweated to it, taper pins about $\frac{3}{16}$ or $\frac{1}{4}$ being used, according to size of tubes, the portion of socket outside the tube having previously been milled to some suitable form, either forked or plain. Such a socket with a plain end would be as illustrated in Fig. 113.

Another method (Fig. 114) is to make a saw-cut for a short distance down the tube, the two portions being then opened out to receive a reinforcing U piece, which is then either brazed in or edge welded. The U piece is generally made of a somewhat thicker gauge than the tube itself. This method of "ending" tubes is still very popular and gives satisfaction ; it is, however, much more expensive than the method already described, in which a reinforcing piece of tube is inserted at each end of the tube, which is then flattened and dip-brazed.

Landing Gears

Landing gear legs are generally made either of round or streamline tubes, and the load is always compressive, so that it is usual to fit a turned socket into the tube end, as described above. It should be made certain that the pins in the sockets are strong enough to withstand the heavy loads taken on landing.

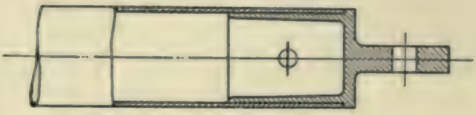


FIG. 113.—SOCKET END FOR TUBE

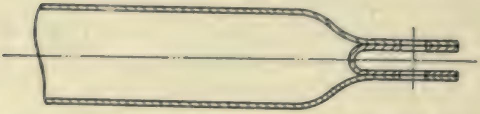


FIG. 114.—TUBE REINFORCEMENT WITH U PIECE

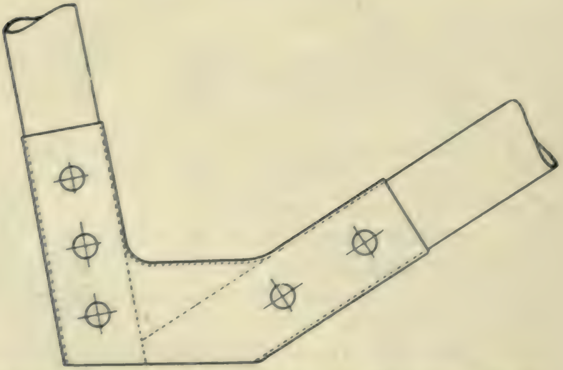


FIG. 115.—BOTTOM BRACKET OF UNDERCARRIAGE

If streamline tubing is used, then a telescopic fitting must be made. In this case, too, when streamline tubing is used, a bracket (Fig. 115) must be designed to connect the front and rear leg, whereas if round tube be used, both front and rear legs may be made from one length of tube.

The operation of bending the tube without local damage is not too easy, and hence, even with round tube, often a short horizontal piece with sockets connects front and rear tubes. (Fig. 116.)

Axle tubing for the landing gear should be of particularly good material, since it must withstand heavy bending-moments due to the overhang of the wheel from the rubber shock absorber cord fixing at the bottom bracket of the legs. On this account the overhang should be kept as small as practicable. The cross-bracing wires are of streamline form.

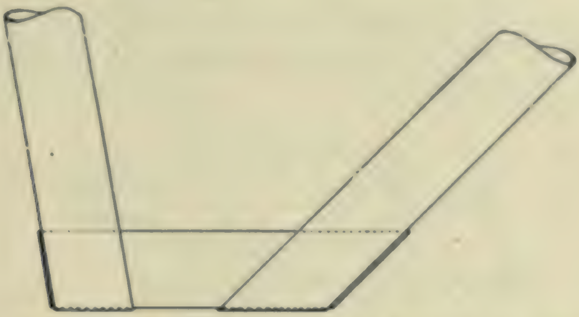


FIG. 116.—BOTTOM BRACKET OF UNDERCARRIAGE

The offset of these wires, taken as they are from the front legs only, should be kept within reasonable limits, if not entirely eliminated. It should also be the endeavour of the designer to avoid offset couples at the attachment of the front and rear legs to the fuselage.

Tail Plane and Elevators

As far as regards detail design in the tail plane and elevators, almost everything already discussed in connection with main planes applies equally well here and there is no point in duplicating the work.

Oftentimes, instead of the usual streamline wire external bracing, streamline tubes are used either above or below the unit, and completely brace it, since they are capable of withstanding either tension or compression. This ought to be done very judiciously or else it will be found that the weight becomes excessive. Stiffer fittings would necessarily be required in the case of struts rather than wires. The attachment of the plane to the rear portion of the fuselage ought also to receive due consideration, especially where tail plane adjusting gear is fitted.

Hollow Wooden Struts

Hollow wooden lengths suitable for struts, etc., have been designed with considerable success. They possess the properties of strength and lightness in good degree. Notable amongst such designs are the productions of the McGruer Hollow Spar Co., Ltd., who have produced large quantities of their patent bent-wood spars for use on aeroplanes during the war. Tests made upon such have been quite satisfactory, as will be seen from the results given in Table 28, for which we are indebted

TABLE 28

TESTS ON MCGRUER HOLLOW SPRUCE SPARS

Royal Aircraft Establishment.

	Length. ins.	Ex. dia. ins.	T. ins.	Load. lbs.	Weight.	Load per sq. in.
1	19·625	1·08	·22	2600	2¾ oz.	4420 lbs.
2	25·25	1·27	·26	3120	5 ,,	3770 ,,
3	39·0	1·58	·26	3225	10¾ ,,	3015 ,,
4	55·625	1·82	·32	3310	22½ ,,	2205 ,,
5	21·0	1·52	·27	5540	5⅞ ,,	5250 ,,

Lieutenant Scobel, London.

	Length. ins.	Ex. dia. ins.	T. ins.	Load. lbs.	Weight.	Load per sq. in.
1	62	3·49	·299	10300	3 lbs. 4 ozs.	3440 lbs.
2	67·12	3·45	·301	10990	3 ,, 8½ ,,	3700 ,,
3	38	1·572	·22	4800	9 ,, 8 ,,	4420 ,,
4	67·75	2·40	·284	4000	2 ,, 1 ,,	2125 ,,
5	113·37	3·51	·31	4360	5 ,, 15 ,,	1400 ,,

Professor Lea, Birmingham.

	Length. ins.	Ex. dia. ins.	T. ins.	Load. lbs.	Weight.	Load per sq. in.
1	68	3·43	·37	12650	3 lbs. 5 ozs.	3570 lbs.
2	124	3·25	·375	4630	6 ,, 3 ,,	1180 ,,
3	64	3·2	·35	11800	2 ,, 13 ,,	3750 ,,
4	100	3·2	·35	5300	4 ,, 8 ,,	1690 ,,
5	124	4·48	·26	7400	7 ,, 6 ,,	2125 ,,

"T" is thickness of wall. All these spars were circular.

to the McGruer Hollow Spar Co. The cross section of the McGruer strut is shown in Fig. 117, from which it will be

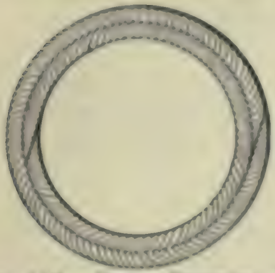


FIG. 117.—CROSS SECTION OF McGRUER CIRCULAR STRUT

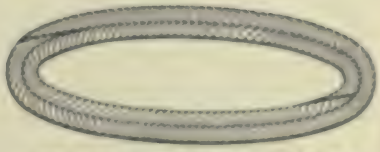


FIG. 118.—CROSS SECTION OF McGRUER OVAL STRUT

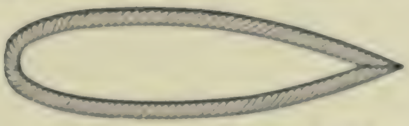
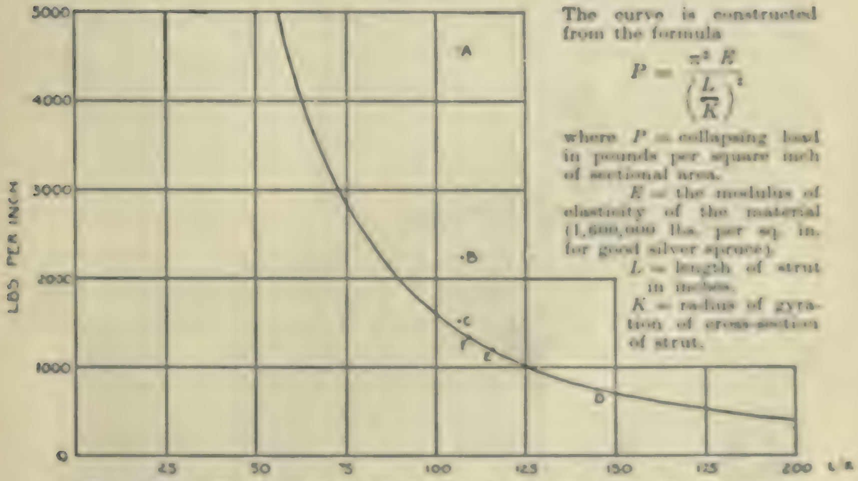


FIG. 119.—CROSS SECTION OF McGRUER STREAMLINE STRUT



The curve is constructed from the formula

$$P = \frac{\pi^2 E}{\left(\frac{L}{K}\right)^2}$$

where P = collapsing load in pounds per square inch of sectional area.

E = the modulus of elasticity of the material (1,600,000 lbs. per sq. in. for good silver spruce).

L = length of strut in inches.

K = radius of gyration of cross-section of strut.

FIG. 120.—RESULTS OF McGRUER TESTS ON SPRUCE STRUTS PLOTTED WITH REFERENCE TO THE EULER CURVE

observed that the thin boards from which the strut is made are bent breadth-wise, the edges being joined in the form of a scarf joint with waterproof glue. As will be seen from Figs. 118 and 119, the cross sectional shape is not confined to the circular form, but that oval and streamline shapes are also possible ; furthermore the struts can also be made with taper or merely parallel. The Euler curve of loads is shown in Fig. 120, and a series of points representing actual tests on long spars are plotted relative to the curve. Point *A* represents the result of a strut test between parallel plates ; Point *B* represents tests of a similar strut between small balls in sockets; Point *C* represents a test between small balls without sockets. The end fittings for such struts are important when used on aeroplanes, and probably the best way is to plug the ends with solid pieces of wood in order that a good attachment may be obtained with the metal fitting usually forming the connection of struts to the supported object.

There are other forms of strut made by wrapping very thin strips of wood spirally, good quality glue being used.

It is doubtful, however, if these may be used with the same success as struts made in other ways.

The old method, and it is used to some extent now, of manufacturing hollow struts is to spindle out the middle portions of two halves, which are then carefully glued together. This method is used in the construction of aileron and elevator spars also. A tongue is inserted along each of the two edges to be glued together, in order to increase the glued area. These are satisfactory, but not cheaply made.

Welding, Brazing, and Sweating

Joints made by any of these processes may not always be assumed to be 100% efficient (*i.e.*, the finished job is not taken to be as strong as if it were made for a single piece of material).

Discretion must in every case be used in deciding what " efficiency factor " shall be given to a particular case.

WELDING should always be done with care because the temperature of the flame is so high that it is comparatively easy to locally over-heat the metal ; it is also a fairly expensive process. It is, of course, a particularly convenient process to be able to make use of, but in general, unless the operator is really skilled, it should be avoided except in places where strength is relatively unimportant. It is advisable in many cases to anneal after welding because local brittleness often shows itself in the vicinity of the weld.

BRAZING is another process used in the connecting of tubes, plates, and fittings generally. It may be done either by hand (*i.e.*, by means of a blow-pipe) or in a dipping bath.

There is not a great deal of danger from over-heating when done by hand, but unless the dipping bath be carefully regulated in temperature, it has been found that over-heating does occur and serious cracking of the plates takes place. If parts are annealed after brazing, then care must be taken that the melting point of the spelter is never approached, otherwise, of course, it will simply run out of the joint.

SWEATING consists of soldering the surfaces to be joined together. The surfaces are gently heated until the solder between them fuses. On cooling, the parts concerned are joined together. The temperature required for this operation is low compared with that required for welding or brazing. In the case of the two latter the parts concerned are annealed sometimes to a serious extent. A high-class carbon steel, for instance, would have its yield point reduced from, say, 30 tons per square inch to about half that value. Therefore, by soldering, owing to the comparatively low temperature the metal is not annealed to anything like the same extent, and this is an obvious advantage.

Different qualities of solder may be used, some having lower melting points than others. When these latter are used, the method is known as soft-soldering.

Regarding the relative strengths of the three methods, welding is strongest, with brazing taking second place.

Deflection of Spars, Etc.

It is probable that for some reason the deflection of a spar is desired at some point or other. For all practical purposes the graphical method of performing the double integration is much to be preferred. As an example, take the spar* whose B.M. diagram is drawn in Fig. 121, it is required to find the deflection at every point due to the loading from which this B.M. diagram was produced. To obtain the deflection, a curve of slope must be drawn, and from this curve a deflection diagram is deduced in a similar manner. The graphical method

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* This spar has been chosen because it is supported at more points than the spars considered in Chapter II *et seq.*

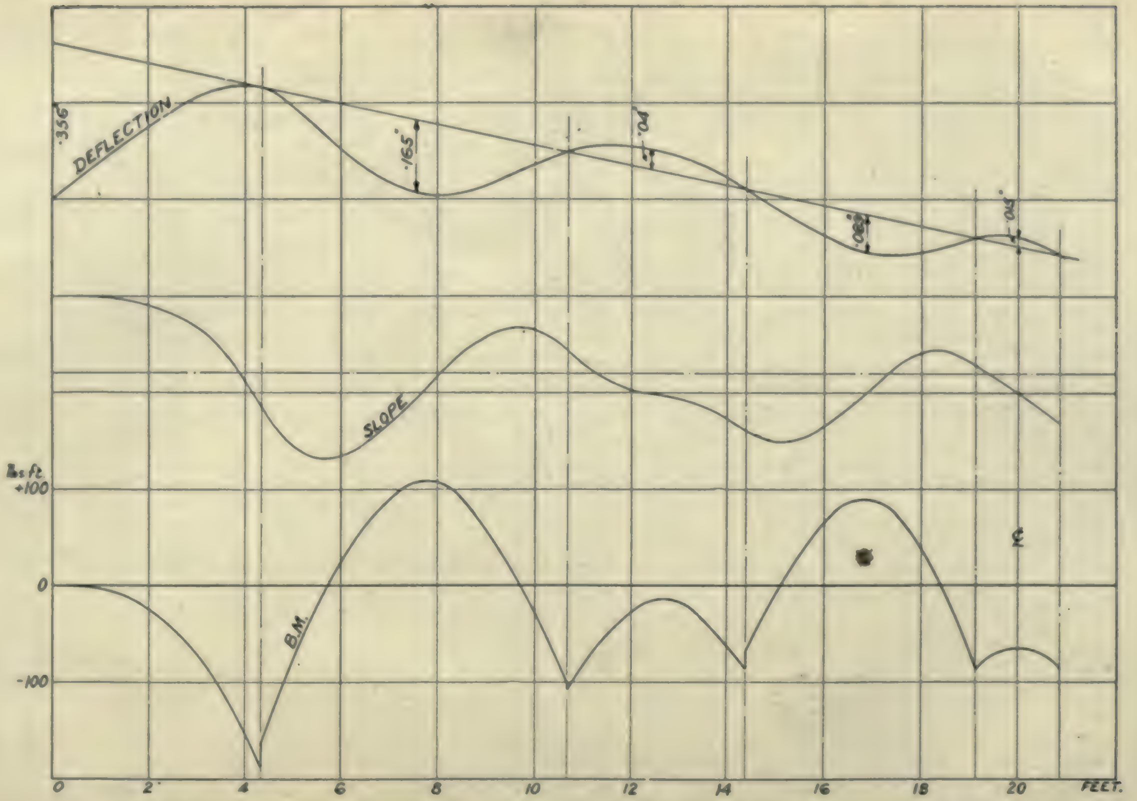

FIG. 121.—BENDING-MOMENT AND DEFLECTION DIAGRAM FOR WING SPAR

is more easily performed if the bending-moment diagram be drawn on squared paper, then, commencing from either end, begin the integration of the B.M. diagram, making the ordinate at any point of the curve of slope equal to the area under the B.M. curve up to that point from the end considered.

The area is obtained by merely counting of squares, which may be done quite quickly ; it must be done carefully, however, otherwise inaccuracies will show themselves in the deflection diagram. Having drawn the curve of slope, a suitable base line, preferably a mean line, should now be drawn and the process of integration repeated to produce the deflection curve. A check on the accuracy of the integration is found, inasmuch as the base of the deflection curve ought to pass through the supports at points which are collinear (*see* Fig. 121), deflections being measured from this line.

All that remains is that the scales be worked out as follows—

Linear scale 1 in. = 2 ft.
B.M. scale 1 in. = 100 lbs. ft.

The curve of slope has been drawn such that 1 in. ordinate = 1 sq. in. B.M. area, and deflection curve has been drawn such that 1 in. ordinate = 1 sq. in. slope curve area.

Hence, 1 in. ordinate of deflection curve

$$= \frac{2 \times 2 \times 100 \times (12)^3}{E \times I} \text{ ins.}$$

In this case $E = 1{\cdot}4 \times 10^6$ lbs. per sq. in.
$I = 2{\cdot}23$ ins.^{4}

$$\therefore \; 1 \text{ in.} = \frac{400 \times 1728}{1{\cdot}4 \times 10^6 \times 2{\cdot}23} = 0{\cdot}22 \text{ in. actual deflection}$$

APPENDIX

WING SPAR STRESSES*

By H. A. Webb, M.A., A.F.Ae.S. and
H. H. Thorne, B.A., B.Sc.

(1) The calculation of the stresses in the spars of aeroplane wings is important and difficult. The graphical methods commonly used give good results for normal horizontal flight, but are inaccurate when any bays of the spar have large compressive end-loads; and this often happens when the wings are heavily loaded, as in diving or stunting. Hence, these methods are unreliable for the very important purpose of estimating the factor of safety. On the other hand, the exact mathematical solution involves trigonometrical functions in a complicated way. This is, perhaps, not a great disadvantage if the object is merely to check the strength of a wing that is already completely designed. The drawback for the designer is that the physical meaning of each term in the formulae is not clear, so that it is impossible to judge from the formulae the effect that a small modification of the design is likely to have on the strength; each case considered has to be worked through from beginning to end independently.

The object of the following is to give a method free from both these defects. The formulae in 3 and 4 are so simple that the physical meaning of each term is seen at once; and by means of them the bending-moment diagram of the spar can actually be drawn more quickly than by the old-established graphical methods, all the necessary calculations being performed with a slide-rule.

It is true that none of our formulae are mathematically accurate but they are approximate enough for practical purposes for all loads, from those of normal flight up to those that overstress the wing, and they can be used with advantage to find the factor of safety.

Mathematical readers can verify the closeness of the approximation by term-by-term comparison with the exact mathematical formulae.

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* First published in *Aeronautics*, Jan., 1919.

(2) Notation

To fix ideas, consider the left wing only The air load in normal flight is positive. Hogging bending-moments, such as those due to the upward air load in normal flight, are positive; sagging bending-moments are negative. We make the usual assumptions that each spar is a continuous beam whose supports are collinear; that the cross-section of each bay is uniform, though it may be different in different bays; and that the load due to air forces is uniformly distributed along the bay, though it may be different in different bays.

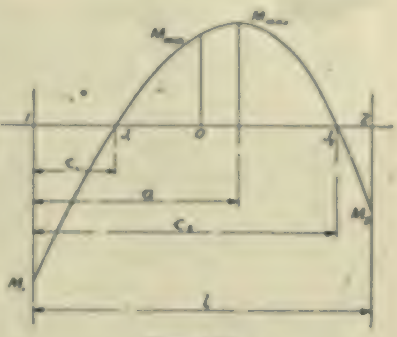

FIG. 122

For any bay 1, 2 (*see* Fig. 122).

l feet = the length of the bay
I in units = the cross-sectional moment of inertia in bending
A sq in. = the sectional area
E lb. per sq. in. = Young's modulus of elasticity.
w lb. per ft. = the uniform upward load on the bay, due to air forces.
M_1 lb. ft. = the fixing moment at 1.
M_2 lb. ft. = the fixing moment at 2.
M_{mid} lb. ft. = the bending-moment at the middle of point O of the bay.
M_{max} lb. ft. = the maximum bending-moment in the bay occurring at
a ft. = the distance from the left end of the bay.
c_1, c_2 ft. = the distances from the left end of the bay of the points of inflexion J_1 and J_2, where the bending-moment is zero.

It should be noticed that M_1 and M_2 are generally negative, and consequently M_{max}, although it represents the highest point of the bending-moment diagram in the bay, is not necessarily the greatest bending-moment, because it may be, as in c_2, numerically less than M_1 or M_2, or both; and also that, though there are usually two points of inflexion, J_1 and J_2, there may be only one.

The fixing moments on the two sides of a support are often unequal, because of the offset pull of the wires.

When this happens—

M_{2_L} lb. ft. = the fixing moment to the left of 2.
M_{2_R} lb. ft. = the fixing moment to the right of 2.
P lb. = the direct compressive or tensile end-load in the bay.
P_e lb. = The crippling end-load of the bay regarded as a pin-jointed strut, according to Euler's formula, so that

$$P_e = \frac{\pi^2 EI \times 144}{l^2}, \quad (l \text{ in ft.})$$

We assume that $P < P_e$; this may seem obvious, but it has been shown that, as a matter of pure theory, a bay may, in certain circumstances, be held up by the neighbouring bays to support more than its individual crippling end-load, P_e, but it would be rash to rely on this in design.

The formulae from which the bending-moment diagram is drawn depend on whether the spar is in compression or tension. In a biplane in normal flight, the spars of the top wing are in compression, and those of the bottom wing in tension.

Spar in Compression

Theorem of three moments for two consecutive bays 1–2 and 2–3 is (*see* Fig. 123)—

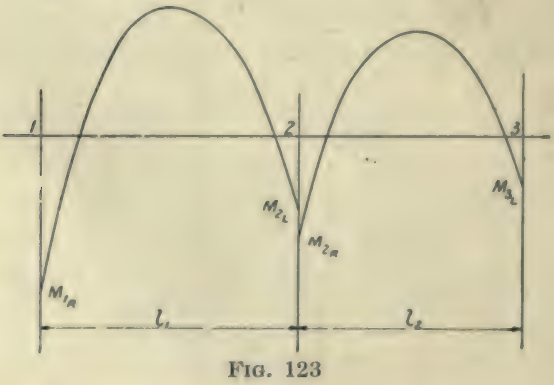

FIG. 123

$$\begin{aligned}&\frac{1}{(P_{e_1}-P_1)l_1}\left\{M_{1_R}\left[1+\cdot2\frac{P_1}{P_{e_1}}\right]+2M_{2_L}\left[1-\cdot38\frac{P_1}{P_{e_1}}\right]\right.\\&\left.+\tfrac{1}{4}w_1 l_1^2\left[1-\frac{1}{70}\frac{P_1}{P_{e_1}}\right]\right\}+\frac{1}{(P_{e_2}-P_2)l_2}\left\{M_{3_L}\left[1+\cdot2\frac{P_2}{P_{e_2}}\right]\right.\\&\left.+2M_{2_R}\left[1-\cdot38\frac{P_2}{P_{e_2}}\right]+\tfrac{1}{4}w_2 l^2_2\left[1-\frac{1}{70}\frac{P_2}{P_{e_2}}\right]\right\}=O\ . \qquad (1)\end{aligned}$$

For any bay 1–2 (Fig. 122)—

$$M_{mid}=\frac{P_e}{P_e-P}\left\{\tfrac{1}{2}(M_1+M_2)\left[1+\cdot26\frac{P}{P_e}\right]+1\cdot02\frac{wl^2}{8}\right\} \qquad (2)$$

$$M_{max}=M_{mid}+\tfrac{1}{2}\frac{(M_2-M_1)^2}{wl^2}\ .\quad .\quad .\quad .\quad . \qquad (3)$$

occurring at distance a from left-hand support where

$$a=\tfrac{1}{2}l+\frac{M_2-M_1}{wl}\ .\quad .\quad .\quad .\quad .\quad . \qquad (4)$$

and position of J_1 and J_2 are formed from

$$C_1=\tfrac{1}{2}l+\cdot9\left(\frac{M_2-M_1}{wl}\right)-1\cdot2\sqrt{\frac{M_{max}}{w}}\ .\quad . \qquad (5a)$$

$$C_2=\tfrac{1}{2}l+\cdot9\left(\frac{M_2-M_1}{wl}\right)+1\cdot2\sqrt{\frac{M_{max}}{w}}\ .\quad . \qquad (5b)$$

Spar in Tension

Fig. 123 same as before, then the theorem of three moments for any two consecutive bays 1–2, and 2–3, is—

$$\begin{aligned}&\frac{1}{(P_{e_1}+P_1)l_1}\left\{M_{1_R}\left[1-\cdot12\frac{P_1}{P_{e_1}}\right]+2M_{2_L}\left[1+\cdot32\frac{P_1}{P_{e_1}}\right]\right.\\&\left.+\tfrac{1}{4}w_1 l_1^2\left[1+\frac{1}{80}\frac{P_1}{P_{e_1}}\right]\right\}+\frac{1}{(P_{e_2}+P_2)l_2}\left\{M_{3_L}\left[1-\cdot12\frac{P_2}{P_{e_2}}\right]\right.\\&\left.+2M_{2_R}\left[1+\cdot32\frac{P_2}{P_{e_2}}\right]+\tfrac{1}{4}w_2 l^2_2\left[1+\frac{P_2}{80P_{e_2}}\right]\right\}=O. \qquad (6)\end{aligned}$$

for any bay (Fig. 122)

$$M_{mid} = \frac{P_e}{P_e + P}\left\{\tfrac{1}{2}(M_1 + M_2)\left[1 - \cdot 21\,\frac{P}{P_e}\right] + \cdot 98\,\frac{wl^2}{8}\right\} \quad (7)$$

$$M_{max} = M_{mid} + \tfrac{1}{2}\,\frac{(M_1 + M_2)^2}{wl^2} \quad . \quad . \quad . \quad . \quad . \quad (8)$$

at a distance a from the left-hand support, where

$$a = \tfrac{1}{2}\,l + \frac{(M_2 - M_1)}{wl} \quad . \quad . \quad . \quad . \quad . \quad . \quad (9)$$

The formulae for tension are more accurate than the corresponding formulae for spar in compression.

Remarks on the Formulae

The method of using these formulae is as follows: The fixing moments are found by applying (1) to each pair of consecutive bays, in a similar way to the application of the three moment theroem to a continuous beam. The presence of end-load, in this method, does not increase the difficulty of the numerical work.

The formulae (2)–(5) are then used to find in each bay at least two, and usually four, additional points on the curve, including the highest point; the complete bending-moment diagram is then drawn at once by putting in a fair curve through the points found in each bay.

It is very important to notice that the values of the fixing moments found from (1) are usually negative and that these negative signs must be retained in the formulae (2)–(5). For instance, if AB is the bay next to the wing tip, M_{1_L} will be the fixing moment due to the air load on the wing tip, and from our convention of signs it is clear that M_{1_L} is a negative quantity.

The points of inflexion, J_1 and J_2 (Fig. 122), should be found from (5) so that a good curve may be drawn, and also to help the designer to decide where spindling of the spar section should cease. It may be remarked that where M_{max} is negative, J_1 and J_2 do not exist. This is confirmed by (5) because the square root of a negative quantity is imaginary. The numerical work of finding the fixing moments from (1) should always be checked. The formulae (2)–(5), however, check themselves, since the points should lie on a regular curve in any particular bay.

LOAD FACTORS FOR COMMERCIAL AIRCRAFT

The following communiqué—being a report of the Advisory Committee for Aeronautics—has been issued by the Air Ministry—

Introduction

When the employment of aircraft for civil purposes became practicable, owing to the termination of hostilities, representatives of the aircraft industry pointed out the necessity for arriving at an agreement as to the load factors to be worked to in designing commercial aircraft.

Although knowledge on this subject had increased very largely during the war, the bases and methods upon which designers worked varied considerably, and it was obvious that the matter was one requiring careful technical investigation.

The Air Council therefore requested the Advisory Committee for Aeronautics to appoint a Sub-Committee—upon which the various interested bodies should be represented—to consider and report as to the definite rules which should be adopted to govern the load factors required for all classes of civil aircraft, and upon which Certificates of Airworthiness may be based.

The unanimous report of this Sub-Committee, which has now been approved, is given below. Its provisions will be made applicable to designs submitted to the Director of Research after 15th March, 1920.

The memorandum will appear in due course as Report No. T.1425 of the Advisory Committee for Aeronautics.

SCHEDULE OF LOAD FACTORS FOR HEAVIER-THAN-AIR CRAFT, 6th JANUARY, 1920

Heavier-than-air craft are now used for a variety of purposes, including, on the one hand, purely commercial work, and, on the other, stunt flying. The strength necessary for safety is different in the two classes: for the former class the specification need not be so exacting as for the latter in order to ensure adequate safety; further, a lower factor is commercially desirable, since an increase of load factor involves a decrease in the range and load-carrying capacity. Heavier-than-air craft have, therefore, been divided into two classes, having regard to the different qualities of airworthiness required.

This separation will necessitate the granting of two distinct types of certificates; it has been suggested that these should be readily distinguishable by the use of different colours or other means. The aircraft in the "General" Class, including all those not in the "Commercial" Class, should be sufficiently strong to allow of stunt flying of all descriptions. The other class, "Commercial," will include craft used for strictly commercial work, involving only straight flying; stunting of any kind will be prohibited. The schedule on pp. 245 and 246 is given in tabular form under these two heads.

In preparing this schedule the Load Factor Sub-Committee of the Advisory Committee for Aeronautics considered only future designs; it is not intended that the schedule should be made applicable to machines already approved or designs submitted prior to some future date to be fixed by the Air Ministry. Further revision of the present schedule from time to time will doubtless be necessary in order that it may remain in accordance with the demands arising from improvements in the constructional methods and design.

Method of Specifying Strength

The first specification (*a*) determines the load factor when the centre of pressure is in its most forward position: this gives in each case a definite load for every part of the machine, including the tail plane and the fuselage.

The second specification (*b*) gives the load factor when the centre of pressure is in the position corresponding to maximum horizontal speed at ground level: this also gives a definite loading.

The next specification (*c*) gives the factor of safety in a terminal nose dive, a condition which is again quite definite for each aircraft, the airscrews being assumed to be removed.

The above are considered sufficient to determine the strength of the wings and of the tail plane; also for certain loadings of the fuselage when in flight.

For the strength of the fin and rudder, specification (*d*) gives the value of an assumed lift co-efficient, which, when the maximum speed of the craft is known, determines uniquely the strength required in these parts, and also in so far as the lateral loadings are concerned in the fuselage: the values assumed for the lift coefficient are considered sufficient to cover adequately the maximum forces which can be imposed on the rudder and fin by the pilot when exerting his full force so as to obtain as rapid a turn as possible.

Strength of undercarriages, specification (*e*), is dealt with under the conditions of both static and dynamic loads, neither case being considered sufficient by itself to meet all requirements.

For the use of the tables it should be noted that the two terms, factor of safety and load factor, are defined in the accepted engineering sense, *i.e.*—

$$\text{Factor of safety} = \frac{\text{Failing strength of a member of the structure}}{\text{Worst possible load occurring under any condition of flight.}}$$

$$\text{Load factor} = \frac{\text{Failing strength of a member of the structure}}{\text{Load in member under horizontal steady flight conditions.}}$$

COMMERCIAL CLASS

	TOTAL WEIGHT OF AIRCRAFT.			
	Up to 5,000 lbs.	5,000–10,000 lbs.	10,000–30,000lbs	Above 30,000lbs
(*a*) Load factor with C.P. in its most forward position .	6†	6†–5*†	5†–4*	4
(*b*) Load factor with C.P. in the position corresponding to maximum horizontal speed at ground level . . .	4·5†	4·5†–3·75*†	3·75†–3	3
(*c*) *Factor of safety* in terminal nose dive	1·75†	1·75†	1·75†	1·75†
(*d*) Specified lift coefficient for fins and rudders. (Under this loading the factor of safety of the fuselage should be unity)	0·5	0·5	0·5	0·5
(*e*1) Static load factor on undercarriages	6	6–5*	5–4*	4
(*e*2) Specified vertical velocity (ft./sec.) for determining travel of undercarriages .	10	10	10	10

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* The decrease in load factor from the larger value is directly proportional to the increase in the weight of the craft.

† In the case of aircraft which are longitudinally stable over the whole flying range, these figures may be reduced by 0·5.

Schedule of Load Factors and Factors of Safety

GENERAL CLASS

	TOTAL WEIGHT OF AIRCRAFT.		
	Up to 3,000 lbs	3,000–10,000 lbs.	Above 10,000 lbs.
(*a*) Load factor with C.P. in most forward position	8	8–6*	6
(*b*) Load factor with C.P. in the position corresponding to maximum horizontal speed at ground level	6	6–4·5*	4·5
(*c*) *Factor of safety* in a terminal nose dive	1·75	1·75	1·75
(*d*) Specified lift coefficient for fins and rudders. (Under this loading the factor of safety of the fuselage should be unity)	0·6	0·6	0·6
(*e*1) Static load factor on undercarriages	8	8–6*	6
(*e*2) Specified vertical velocity (ft./sec.) for determining travel of undercarriages	10	10	10

* The decrease in load from the larger value is directly proportional to the increase in the weight of the craft.

In addition to the above, it is essential (*f*) that, in the case of any one flying wire, or duplicate pair of wires, being removed from the aircraft, the strength of the remaining portions should be such that at least one-half of the scheduled load factors and factors of safety shall be obtained under each of the foregoing conditions. For this particular case incidence or other normally redundant bracing is naturally assumed operative where necessary.

Methods of Calculation

In using the above schedules the calculations of strength will be checked by the methods of the *Handbook of Strength Calculations* (H.B. 806, Second Edition), published by the Technical Department, Air Ministry. In all calculations and in specifications of Load Factors and Factors of Safety, design figures should be employed, and not those obtained from breaking tests. Further, when dealing with steel tubes, the specifications of the Engineering Standards Committee, based on the figures for yield points, are approved for stress calculation purposes; for streamline wires the breaking load should be taken in preference to the yield point, which is, in these cases, ill-defined. Redundant wires, except as mentioned in (*f*), are to be considered as neglected.

With a view to facilitating the use of the schedule the following additional matter should be noted—

(1) That for (*a*) and (*b*) the positions of the centre of pressure should preferably be determined by a test on a model with a 6 in. chord at 60 ft./sec., but that the Air Ministry might agree to issue a certificate based on results for a monoplane model of 3 in. chord tested at 40 ft./sec. For biplanes, a biplane test is preferred ; similarly for triplanes.

For case (*b*) the maximum horizontal speed at ground level should be obtained from the formulae given in section (2) below. From this speed the value of the lift co-efficient is obtained, and the position of the centre of pressure determined from model tests as defined above.

(2) That for (*b*) and (*d*) the maximum speed at ground level should be defined by the following formulae giving the maximum speed for given loadings and weight of aircraft ; should the manufacturer desire to specify the maximum speed in any other manner he is at liberty to make a special application.

FOR SINGLE ENGINE AIRCRAFT.

$$\left(V\sqrt{\frac{\rho}{\omega}} - 1{\cdot}79\right)^2 = 3{\cdot}16\left(N\sqrt{\frac{\rho}{\omega}} - 0{\cdot}316\right)$$

FOR MULTI ENGINE AIRCRAFT.

$$\left(V\sqrt{\frac{\rho}{\omega}} - 1{\cdot}64\right)^2 = 2{\cdot}99\left(N\sqrt{\frac{\rho}{\omega}} - 0{\cdot}310\right)$$

FOR BOAT SEAPLANES.

$$\left(V\sqrt{\frac{\rho}{\omega}} - 1{\cdot}425\right)^2 = 2{\cdot}74\left(N\sqrt{\frac{\rho}{\omega}} - 0{\cdot}304\right)$$

Where N, V, ω and ρ are the power per unit weight, the speed, the loading, and the density in any system of consistent units : *e.g.*, N in ft. lbs. per sec. per lb. ; V in ft./secs. ; ω in lbs./sq. ft. ; and ρ in slugs/cubic ft. (at ground level).

The above formulae may also be expressed in the form—

FOR SINGLE ENGINE AIRCRAFT.

$$\left\{\frac{V}{\sqrt{\omega}} - 25\right\}^2 = 16{\cdot}6\left\{\frac{N}{\sqrt{\omega}} - 11{\cdot}8\right\}$$

For Multi Engine Aircraft.

$$\left\{\sqrt{\frac{V}{\omega}} - 23\right\}^2 = 15{\cdot}7 \left\{\sqrt{\frac{N}{\omega}} - 11{\cdot}6\right\}$$

For Boat Seaplanes.

$$\left\{\sqrt{\frac{V}{\omega}} - 20\right\}^2 = 14{\cdot}4 \left\{\sqrt{\frac{N}{\omega}} - 11{\cdot}35\right\}$$

Where N is horse-power per 1,000 lbs., *i.e.*, $\frac{1000 \text{ H.P.}}{W}$

W being the weight, and H.P. the horse-power of the aircraft in question. V is in miles per hour at ground level ; and ω is wing loading in lbs. per square feet.

(3) That for (*c*), the case of the limiting nose dive calculations submitted by the manufacturers would be considered by the Air Ministry ; otherwise, the following table with the necessary extensions for other wing sections would be employed.

TABLE OF TAIL PLANE LOADS IN LIMITING NOSE DIVE

Type of Aircraft.	Wing Section Employed.	
	R.A.F. 15.	R.A.F. 6C.
Single Engine	$0{\cdot}6\ \frac{W c}{l}$	$0{\cdot}9\ \frac{W c}{l}$
Multi Engine	$0{\cdot}55\ \frac{W c}{l}$	$0{\cdot}7\ \frac{W c}{l}$
Boat Seaplane	$0{\cdot}5\ \frac{W c}{l}$	$0{\cdot}5\ \frac{W c}{l}$

Where W is the weight of the aircraft, c is the chord of the main planes ; and l is the distance from the centre of gravity of the aircraft to the centre of pressure of the tail plane.

(4) That in the case of (*d*) the static strength of the undercarriage should be checked for the four following cases—

(i) When the aircraft lands on an even keel and with the chord of the main planes horizontal.

(ii) When the aircraft lands on an even keel and so that the tail skid and undercarriage touch the ground simultaneously.

(iii) When the aircraft lands so that one wing tip touches the ground at the same time as the main undercarriage, and with the chord of the main planes horizontal.

(iv) When the aircraft lands so that one wing tip touches the ground at the same time as the main undercarriage, and so that the tail skid and undercarriage touch the ground simultaneously.

In (i) and (ii) the load factor should be that specified, and in (iii) and (iv) the load lactor should be at least half that specified.

As regards length of travel of the undercarriage, its amount should be such that the total work of compression is equal to the value of $\frac{1}{2}\frac{wv^2}{g}$ where w is the weight of the machine and v is the specified vertical velocity; also, the maximum load should not exceed the specified static load on the undercarriage.

USEFUL TABULATED DATA

TABLE 29

High Tensile Steel Wire

Gauge.	Sectional area.	Breaking load lbs.
8	0·02011	3600
9	0·01628	2920
10	0·01287	2450
11	0·01057	2130
12	0·00850	1520
13	0·00665	1340
14	0·00503	1070
15	0·00407	868
16	0·00322	720
17	0·00246	550
18	0·00181	405
19	0·00126	282
20	0·00102	251
21	0·00080	200

These breaking loads are obtained on the assumption of stresses varying from 80 tons per square inch to 110 tons per square inch.

TABLE 30

Strength of Swaged and Streamline Wires

Size.	Ultimate strength.
4BA	1050
2BA	1900
$\frac{1}{4}$ in.	3450
$\frac{9}{32}$,,	4650
$\frac{5}{16}$,,	5700
$\frac{11}{32}$,,	7150
$\frac{3}{8}$,,	8500
$\frac{13}{32}$,,	10250
$\frac{7}{16}$,,	11800
$\frac{15}{32}$,,	13800
$\frac{1}{2}$,,	15500
$\frac{9}{16}$,,	20200
$\frac{5}{8}$,,	24700

The ends are threaded with the size in which the wire is known, thus 4*BA* is threaded with No. 4 British Association thread, and $\frac{1}{4}$ is threaded with size $\frac{1}{4}$ in. B.S.F. thread.

STRENGTH OF BOLTS

TABLE 32

British Association and British Standard Fine Threads

Diameter (ins.)		Cross section Area (ins.²).		*MI* for bending (on full diameter), ins.⁴	*Z* for bending (on full diameter) ins.³	Allow'ble B.M. (lbs. ins.) for stress of 80,000 lbs./ins.²	Allow'ble shear load on stress of 40,000 lbs./ins.²	Allow'ble tensile load on stress of 80,000 lbs./ins.²
Full.	Core.	Full diameter	Core diameter B.S.F. thread					
4BA ·142 in.	·11	·0158	·0096	$\cdot1995 \times 10^{-4}$	$\cdot2800 \times 10^{-3}$	22	630	780
2BA ·185 in.	·147	·0269	·0169	$\cdot5770 \times 10^{-4}$	$\cdot6240 \times 10^{-3}$	51	1080	1350
1/4 ..	·1988	·0491	·0316	$\cdot1917 \times 10^{-3}$	$\cdot1534 \times 10^{-2}$	123	1960	2520
5/16 ..	·2544	·0767	·0508	$\cdot4676 \times 10^{-3}$	$\cdot299 \times 10^{-2}$	240	3070	4070
3/8 ..	·3110	·1105	·0760	$\cdot9707 \times 10^{-3}$	$\cdot5177 \times 10^{-2}$	414	4430	6080
7/16 ..	·3664	·1503	·1054	$\cdot1800 \times 10^{-2}$	$\cdot8220 \times 10^{-2}$	658	6000	8430
1/2 ..	·4200	·1963	·1385	$\cdot3068 \times 10^{-2}$	$\cdot1227 \times 10^{-1}$	982	7850	11100
9/16 ..	·4825	·2485	·1828	$\cdot4920 \times 10^{-2}$	$\cdot1750 \times 10^{-1}$	1400	9920	14600
5/8 ..	·5335	·3068	·2235	$\cdot7490 \times 10^{-2}$	$\cdot2397 \times 10^{-1}$	1920	12250	17900
11/16 ..	·5960	·3712	·2790	$\cdot1094 \times 10^{-1}$	$\cdot3185 \times 10^{-1}$	2550	14850	22300
3/4 ..	·6433	·4418	·3250	$\cdot1553 \times 10^{-1}$	$\cdot4142 \times 10^{-1}$	3320	17700	26000
13/16 ..	·7058	·5185	·3913	$\cdot2140 \times 10^{-1}$	$\cdot5270 \times 10^{-1}$	4210	20700	31300
7/8 ..	·7586	·6013	·4520	$\cdot2877 \times 10^{-1}$	$\cdot6577 \times 10^{-1}$	5260	24000	36200
15/16 ..	—	·6903	·5295	$\cdot3800 \times 10^{-1}$	$\cdot8110 \times 10^{-1}$	6490	27600	42300
1 ..	·8719	·7854	·5971	$\cdot4909 \times 10^{-1}$	$\cdot9817 \times 10^{-1}$	7860	31400	47800
1 1/8 ..	·9827	·9940	·7585	$\cdot7863 \times 10^{-1}$	·1398	11180	39800	60700
1 1/4 ..	1·1077	1·2272	·9637	·1198	·1917	15350	49000	77100
1 3/8 ..	1·2149	1·4849	1·1593	·1755	·2552	20400	59400	92800
1 7/16 ..	—	1·6230	—	·2100	·2920	23400	—	—
1 1/2 ..	1·399	1·7671	1·4100	·2485	·3313	26500	70600	113000
1 9/16 ..	—	1·9175	—	·2940	·3770	30100	—	—
1 5/8 ..	1·4649	2·0739	1·6854	·3431	·4213	33700	82900	135000
1 11/16 ..	—	2·2365	—	·3980	·4710	37700	—	—
1 3/4 ..	1·567	2·4053	1·9285	·4604	·5261	42100	96200	154000
1 13/16 ..	—	2·5802	—	·5310	·5860	46900	—	—
1 7/8 ..	—	2·7612	—	·6067	·6471	51750	—	—
1 15/16 ..	—	2·9483	—	·6900	·7120	56900	—	—
2 ..	1·817	3·1416	2·593	·7854	·7854	62800	125700	207000

TABLE 33

Imperial Standard Wire Gauge

Size on Gauge.	Diameter.	
	Inches.	Millimetres.
7/0 . . .	·500	12·2
6/0 . . .	·464	11·8
5/0 . . .	·432	11·0
4/0 . . .	·400	10·2
3/0 . . .	·372	9·4
2/0 . . .	·348	8·8
1/0 . . .	·324	8·2
1 . . .	·300	7·6
2 . . .	·276	7·0
3 . . .	·252	6·4
4 . . .	·232	5·9
5 . . .	·212	5·4
6 . . .	·192	4·9
7 . . .	·176	4·5
8 . . .	·160	4·1
9 . . .	·144	3·7
10 . . .	·128	3·3
11 . . .	·116	3·0
12 . . .	·104	2·6
13 . . .	·092	2·3
14 . . .	·080	2·0
15 . . .	·072	1·8
16 . . .	·064	1·6
17 . . .	·056	1·4
18 . . .	·048	1·2
19 . . .	·040	1·0
20 . . .	·036	0·9
21 . . .	·032	0·8
22 . . .	·028	0·7
23 . . .	·024	0·6
24 . . .	·022	0·55
25 . . .	·020	0·50
26 . . .	·018	0·45
27 . . .	·0164	0·40

SCHEDULE OF STANDARD SIZES OF 2½ : 1 OVAL STEEL TUBES FOR AIRCRAFT

*Air Board T.*11.

Notes

1. The Areas, Radii of Gyration, and Moduli are calculated for tubes of the minimum thickness (*i.e.*, 0·056 in. for 17 gauge tubes, 0·036 in. for 20 gauge tubes, and 0·025 in. for 22 gauge tubes).

2. The Weights per Foot are calculated for tubes of the maximum thickness (*i.e.*, 0·060 in. for 17 gauge tubes, 0·040 in. for 20 gauge tubes, and 0·029 in. for 22 gauge tubes).

Allowance has been made in each case for the tolerances.

3. The Limiting Bending Loads are the loads which at an overhang of 10 ins. will produce a maximum fibre stress of 28 tons per square inch, which is the minimum yield stress allowed in Specification T.6. The limiting loads are calculated for tubes of the minimum section.

4. Specification.—Oval tubes should be ordered to Specification T.1 or T.6. They are straightened and normalized. They should not be ordered in lengths exceeding 10 ft.

Long tubes for trailing edges and similar purposes can be made straighter if built up of shorter pieces, and should be ordered in lengths exceeding 10 ft.

5. Sockets.—Properly formed socket tubes for all standard ovals can be obtained from the tube makers. These socket tubes are $\frac{1}{16}$ in. larger in each outside diameter than the tubes they are to go over; they will therefore be an easy fit when made 20 gauge thick and a close fit when made 18 gauge thick. Unless specially ordered in stated lengths the socket tubes may be supplied in lengths of about 6 ins.

Satisfactory sockets can be made by pressing round tubes into the proper shape. The inside diameter of the round tube should be three-quarters of the larger diameter B of the oval it has to fit.

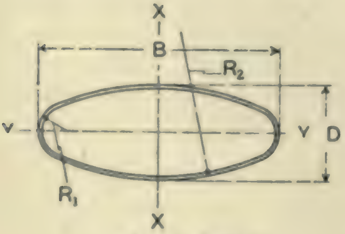
6. Struts.—The curve (Fig. 124) gives the limiting stress, and hence the limiting load, for an oval strut of any size and length, allowing for crookedness and eccentricity of bore permitted in the A.B. Tube Specifications (see *Recent Researches on Tubular Steel Struts*).

TABLE 34

OVAL STEEL TUBES FOR AIRCRAFT

SCHEDULES OF STANDARD SIZES AND PROPERTIES OF SECTIONS

Outside Diameters.		Minimum Area of Section. *See* Note 1.	Weight per Ft. Run of Tube of Maximum Section. *See* Note 2.	Radii of Gyration. *See* Note 1.		Moduli of Section. *See* Note 1.		Limiting Bending Loads at 10 ins. Overhang. *See* Note 3.	
B. Tolerance ± ½ per Cent. of B.	D. Tolerance ± 1 per Cent. of D.			About Axis *XX*.	About Axis *YY*.	About Axis *XX*.	About Axis *YY*.	*Wx*.	*Wy*.
Ins.	Ins.	Sq. Ins.	Lbs.	Ins.	Ins.	Ins.^3	Ins.^3	Lbs.	Lbs.
17 Gauge. Thickness, ·056 in.–·060 in.									
1	·4	·120	·436	·298	·132	·0214	·0105	132	66
1¼	·5	·151	·556	·377	·170	·0344	·0176	216	110
1½	·6	·184	·673	·455	·208	·0513	·0268	321	168
1¾	·7	·217	·791	·534	·246	·0711	·0379	446	238
2	·8	·249	·910	·613	·284	·0943	·0509	592	319
2¼	·9	·282	1·028	·692	·322	·1204	·0659	756	412
2½	1·0	·314	1·147	·771	·361	·1503	·0827	943	518
20 Gauge. Thickness, ·036 in.–·040 in.									
1	·4	·079	·299	·304	·139	·0147	·0077	92	48
1¼	·5	·100	·378	·383	·177	·0236	·0127	148	80
1½	·6	·121	·457	·462	·216	·0345	·0189	217	118
1¾	·7	·142	·536	·541	·254	·0476	·0264	299	165
2	·8	·163	·616	·619	·292	·0628	·0350	394	220
2¼	·9	·184	·695	·698	·330	·0800	·0449	502	282
2½	1·0	·205	·774	·777	·368	·0994	·0562	623	352
Approximately 22 Gauge. Thickness, ·025 in.–029 in.									
⅝	·25	·0341	·134	·189	·086	·0039	·0020	24	13
¾	·3	·0413	·163	·228	·105	·0058	·0031	36	19
1	·4	·0559	·220	·307	·143	·0106	·0058	67	36
1¼	·5	·0704	·278	·386	·181	·0169	·0094	106	59
1½	·6	·0849	·335	·465	·220	·0246	·0138	154	86
1¾	·7	·0996	·392	·544	·258	·0338	·0191	212	119
2	·8	·1140	·450	·623	·296	·0444	·0250	278	156



2½ : 1 Oval.

$$B = 2{\cdot}5\,D \text{ or } D = {\cdot}4\,B.$$

$$R_1 = {\cdot}3\,D \text{ or } {\cdot}12\,B.$$

$$R_2 = 1{\cdot}1\,B \text{ (approx.)}$$

Area of cross section

$$= 1{\cdot}66\,(b + d)t.$$

Radius of gyration about

$$XX = 0{\cdot}317b.$$

Radius of gyration about

$$YY = 0{\cdot}386d.$$

Where $b = B - t$ and $d = D - t$ and $t =$ thickness.

LIMITING STRESS FOR OVAL TUBULAR STEEL STRUTS

With Round or Pin-jointed Ends

Yield Point of Steel . . 28 tons per sq. in.
Young's Modulus, *E* . . 13600 ,, ,,

L = Length of strut in inches.
K = Radius of gyration of cross section of oval about axis *YY*. (*See* Table 34.)

LIMITING LOAD = Limiting Stress × Area of Cross Section.

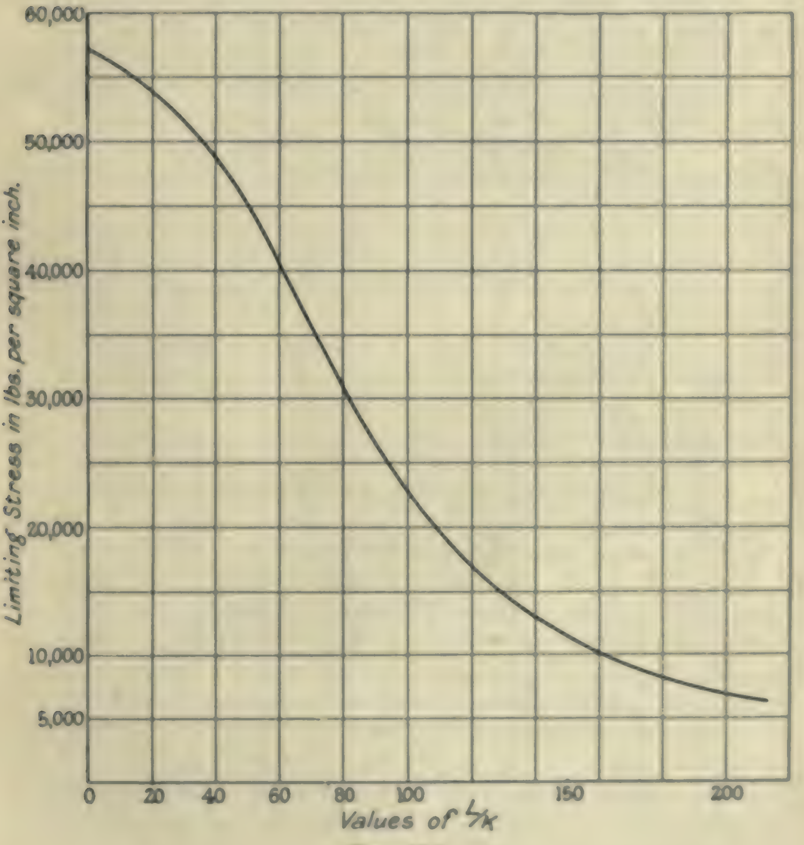


FIG. 124

SCHEDULE OF STANDARD SIZES OF STEEL TUBES FOR AIRCRAFT, OTHER THAN AXLE TUBES

(*Air Board, T* 10.)

Notes

1. The Areas, Moments of Inertia, and Moduli (Bending) are calculated for tubes of the minimum thickness (*i.e.*, 0·056 in. for 17 gauge tubes, 0·036 in. for 20 gauge tubes, and 0·025 in. for 22 gauge tubes). See Table 35.

2. The Weights per Foot are calculated for tubes of the maximum thickness (*i.e.*, 0·060 in. for 17 gauge tubes, 0·040 in. for 20 gauge tubes, and 0·029 in. for 22 gauge tubes). The weight is taken to be 490 lbs. per cubic foot.

3. Telescopic Tubes.—The 17 gauge tubes (omitting the $\frac{5}{16}$ and $\frac{7}{16}$ sizes) form a telescopic series, each fitting over the next size, $\frac{1}{8}$ in. smaller in diameter.

The 22 gauge tubes (with the socket sizes) also form a telescopic series, each fitting over the next size $\frac{1}{16}$ in. smaller in diameter.

4. Tubes for Struts.—Strut tubes are carefully straightened and then blued, but to avoid bending during handling they must not be ordered in lengths exceeding 10 to 12 ft.

5. Tubes for General Purposes.—These are treated in the same way as strut tubes. Long tubes for boundary edges and similar purposes can be made much straighter if built up of shorter pieces, and should be ordered in lengths not exceeding 10 ft.

6. Tubes for Sockets.—The 17 gauge and 22 gauge tubes may be used for sockets (*see* Clause 3), and also tubes of the special standard thicknesses given below. Additional sizes are provided in the table headed "Tubes for Sockets." Tubes for use as sockets are more readily obtainable than strut tubes, because strut tubes have to be carefully straightened and more accurately heat treated; orders should therefore always state when tubes are intended only for sockets. Such tubes must not be used for carrying loads.

Tubes for use as sockets should not be ordered in lengths exceeding 4 ft.; if the lengths they are going to be cut into are specified, the tubes will be supplied in multiples of these lengths; if not, they will be supplied in random lengths of 4 ft. and under.

7. Special Standard Thicknesses.—

No. 1. Special $0{\cdot}090\,{}^{-\,0}_{+\,\cdot007}$ —fits over a tube $\frac{3}{16}$ in. smaller in O.D.

" 2. " $0{\cdot}120\,{}^{-\,0}_{+\,\cdot009}$ —fits over a tube $\frac{1}{4}$ in. smaller in O.D.

" 3. " $0{\cdot}150\,{}^{-\,0}_{+\,\cdot011}$ —fits over a tube $\frac{5}{16}$ in. smaller in O.D.

8. Tolerances and Fits.—The tolerances specified give an average clearance between tube and socket of about 0·008 in.

9. Orders.—Orders for standard tubes must state if they are for *struts* or *sockets*.

If they are for *struts* the order must give for each tube—

(*a*) Number of tubes and length of each, *or* total length and lengths in which it is to be used.

(*b*) Outside diameters and thicknesses or gauges.

(*c*) Specification number, which defines the quality of the steel.

If the tubes are for *sockets* the order must give—

(*d*) Total length required, and when practicable, the lengths the tubes are going to be cut into, so that exact multiples may be supplied.

(*e*) Outside diameters and sizes they are to take.

(*f*) Specification number.

Example.—

Supply 1000 strut tubes 5 ft. 6 ins. long, 1 in. diameter, 17 gauge, *or* Supply 6500 ft. of 1 in. diameter, 17 gauge tube for cutting into 5 ft. 6 in. lengths. Specification, T.6.

Also supply 100 ft. of socket tubing 1⅛ in. O.D. to take 1 in., for cutting into 9 in. pieces. Specification, T.26.

10. Strength of Struts.—Figs. 125, 126 and 127 give the limiting loads for struts of any length made of any of the standard sizes of steel tubes.

Allowance is made for the maximum crookedness and eccentricity of bore which is permitted in the Air Board Specifications.

The *Limiting Load* given by the curves is the load which makes the maximum stress in the tube equal to the yield point (*i.e.*, to 28 tons per square inch).

The *Collapsing Load* of a strut is either the same as, or a little higher than, the limiting load (see *Recent Researches on Tubular Steel Struts*, from which the curves are taken).

TABLE 35

SCHEDULE OF STANDARD SIZES OF STEEL TUBES FOR AIRCRAFT

17 Gauge. Thickness 0·056 inch (Tolerance $^{-\,0}_{+\,0{\cdot}004}$ in.)

Nominal Size. Outside Diameter (Tolerance ± ·004 in.). Ins.	Minimum Area of Section. Ins.^{2}	Maximum Weight per Foot. Lbs.	Moment of Inertia I. Ins.^{4}	Modulus of Section Z. Ins.^{3}	Nominal Size. Outside Diameter (Tolerance ± ·004 in.). Ins.	Minimum Area of Section. Ins.^{2}	Maximum Weight per Foot. Lbs.	Moment of Inertia I. Ins.^{4}	Modulus of Section Z. Ins.^{3}
1/4	·034	·122	·000174	·00139	1 1/8	·188	·683	·0269	·0479
5/16	·045	·162	·000384	·00246	1 1/4	·210	·763	·0375	·0600
3/8	·056	·202	·000736	·00393	1 3/8	·232	·843	·0506	·0735
7/16	·067	·242	·00125	·00568	1 1/2	·254	·923	·0663	·0884
1/2	·078	·282	·00196	·00783	1 5/8	·276	1·00	·0851	·105
5/8	·100	·362	·00409	·0131	1 3/4	·298	1·08	·107	·122
3/4	·122	·442	·00740	·0197	1 7/8	·320	1·16	·132	·142
7/8	·144	·522	·0121	·0278	2	·342	1·24	·162	·162
1	·166	·602	·0186	·0372	2 1/8	·364	1·32	·195	·183

20 Gauge. (Approx. 1 m/m.). Thickness ·036 in. (Tolerance $^{-\,0}_{+\,0{\cdot}004}$ in.)

Nominal Size. Outside Diameter (Tolerance ± ·004 in.). Ins.	Minimum Area of Section. Ins.^{2}	Maximum Weight per Foot. Lbs.	Moment of Inertia I. Ins.^{4}	Modulus of Section Z. Ins.^{3}	Nominal Size. Outside Diameter (Tolerance ± ·004 in.). Ins.	Minimum Area of Section. Ins.^{2}	Maximum Weight per Foot. Lbs.	Moment of Inertia I. Ins.^{4}	Modulus of Section Z. Ins.^{3}
1/4	·0242	·0897	·000142	·00114	1 1/8	·123	·464	·0183	·0325
5/16	·0312	·116	·000305	·00195	1 1/4	·137	·517	·0253	·0405
3/8	·0383	·143	·000557	·00297	1 3/8	·151	·570	·0340	·0494
7/16	·0454	·170	·000923	·00422	1 1/2	·166	·624	·0444	·0592
1/2	·0525	·197	·00142	·00568	1 5/8	·180	·677	·0567	·0698
5/8	·0666	·250	·00290	·00928	1 3/4	·194	·731	·0712	·0814
3/4	·0808	·303	·00516	·0138	1 7/8	·208	·784	·0880	·0938
7/8	·0949	·357	·00836	·0191	2	·222	·837	·107	·107
1	·109	·410	·0127	·0254	2 1/8	·236	·891	·129	·121

Approximately 22 Gauge. Thickness ·025 in. (Tolerance $^{-\,0}_{+\,0{\cdot}004}$ in.)

Nominal Size. Outside Diameter (Tolerance ± ·004 in.). Ins.	Minimum Area of Section. Ins.^{2}	Maximum Weight per Foot. Lbs.	Moment of Inertia I. Ins.^{4}	Modulus of Section Z. Ins.^{3}	Nominal Size. Outside Diameter (Tolerance ± ·004 in.). Ins.	Minimum Area of Section. Ins.^{2}	Maximum Weight per Foot. Lbs.	Moment of Inertia I. Ins.^{4}	Modulus of Section Z. Ins.^{3}
1/4	·0177	·0685	·000113	·000905	1 1/8	·0864	·339	·0131	·0233
5/16	·0226	·0878	·000234	·00150	1 1/4	·0962	·378	·0181	·0289
3/8	·0275	·107	·000423	·00226	1 3/8	·106	·417	·0242	·0351
7/16	·0324	·127	·000693	·00317	1 1/2	·116	·456	·0315	·0420
1/2	·0373	·146	·00106	·00422	1 5/8	·126	·494	·0403	·0495
5/8	·0471	·185	·00212	·00680	1 3/4	·135	·533	·0504	·0576
3/4	·0569	·223	·00375	·00999	1 7/8	·145	·572	·0622	·0663
7/8	·0668	·262	·00605	·0138	2	·155	·610	·0759	·0756
1	·0766	·301	·00911	·0182	2 1/8	·165	·649	·0909	·0855

EXTRA TUBES FOR SOCKETS ONLY

Approximately 22 Gauge. Thickness 0·025 in. (Tolerance $^{-\,0}_{+\,0{\cdot}004}$ in.)

Outside Diam. Ins.	To take Ins.	Min. Area. Ins.^{2}	Max. Weight per Ft. Lbs.	Outside Diam. Ins.	To take Ins.	Min. Area. Ins.^{2}	Max. Weight per Ft. Lbs.
9/16	1/2	0·042	0·166	1 7/16	1 3/8	0·111	0·437
11/16	5/8	0·052	0·204	1 9/16	1 1/2	0·120	0·460
13/16	3/4	0·062	0·244	1 11/16	1 5/8	0·130	0·515
15/16	7/8	0·071	0·282	1 13/16	1 3/4	0·140	0·554
1 1/16	1	0·081	0·321	1 15/16	1 7/8	0·150	0·592
1 3/16	1 1/8	0·091	0·360	2 1/16	2	0·160	0·631
1 5/16	1 1/4	0·102	0·405				

LIMITING LOADS FOR TUBULAR STEEL STRUTS

Tubes 17 *Gauge Thick*

For tubes of standard dimensions the curves are drawn for tubes of the minimum thickness allowed for this gauge, viz., ·056 in.

Young's Modulus . . .	13600 tons per sq. in.
Yield Point . . .	28 ,, ,,
Air Board Specification .	*T*.6.

$$\text{Equivalent eccentricity of loading} = \frac{\text{Int. Diam.}}{40} + \frac{\text{Length}}{600}$$

Ends, round or pin-jointed.

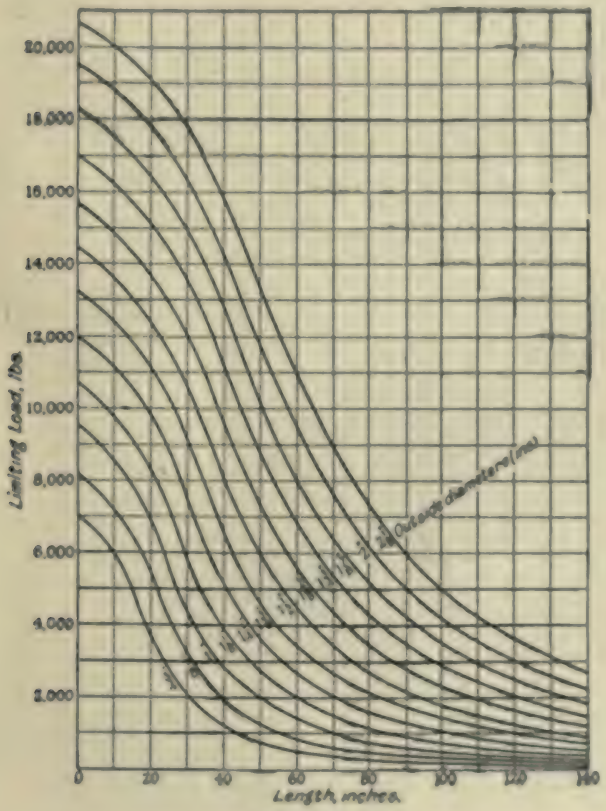


FIG. 125

LIMITING LOADS FOR TUBULAR STEEL STRUTS

20 *Gauge Thick*

For tubes of standard dimensions (Air Board Schedule *T*.10) the curves are drawn for tubes of the minimum thickness allowed for this gauge, viz., ·036 in.

Young's Modulus	. .	13600 tons per sq. in.
Yield Point	. . .	28 " "
Air Board Specification	.	*T*.6.

$$\text{Equivalent eccentricity of loading} = \frac{\text{Int. Diam.}}{40} \times \frac{\text{Length}}{600}$$

Ends round or pin-jointed.

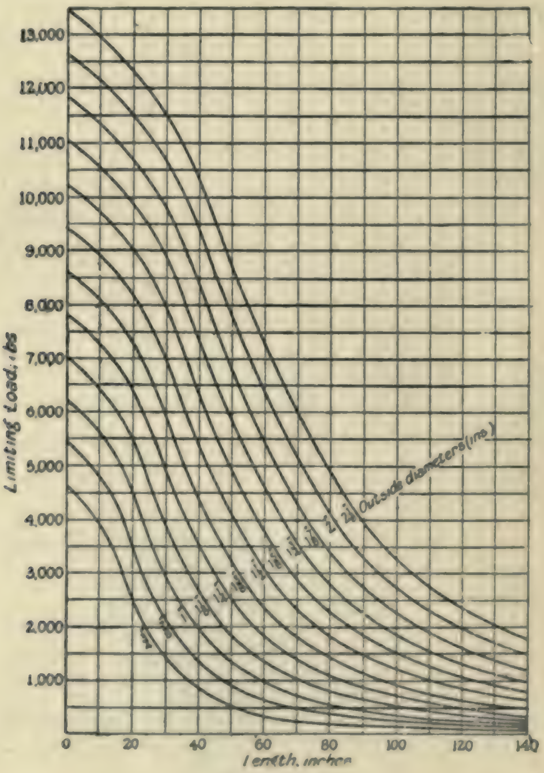


FIG. 126

LIMITING LOADS FOR TUBULAR STEEL STRUTS

Tubes approximately 22 Gauge Thick.

For tubes of standard dimensions the curves are drawn for tubes of the minimum thickness allowed for this gauge, viz., ·025 in.

Young's Modulus . .	13600 tons per sq. in.
Yield Point . . .	28 " "
Air Board Specification .	*T*.6.

$$\text{Equivalent eccentricity of loading} = \frac{\text{Int. Diam.}}{40} + \frac{\text{Length}}{600}$$

Ends round or pin-jointed.

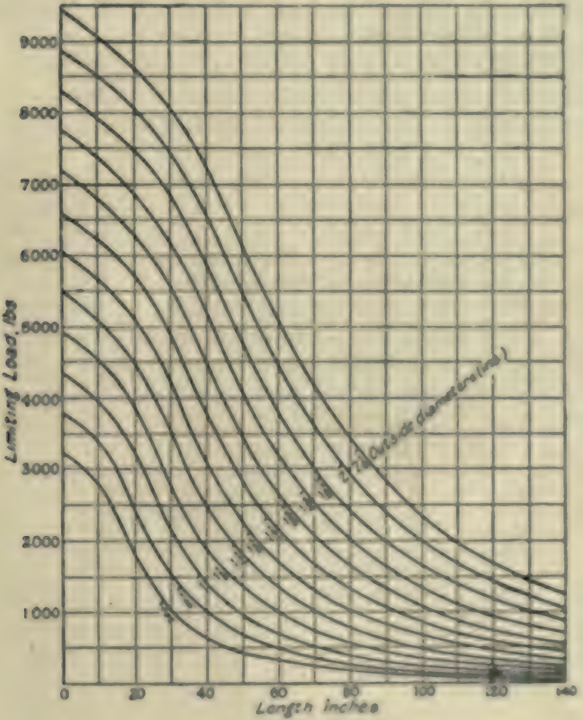


FIG. 127

SUPPLEMENTARY SCHEDULE OF LARGER STANDARD SIZES OF ROUND STEEL TUBES FOR AIRCRAFT

(*Air Board, T.*10. *Supplement*)

Notes

ORDERS.—Tubes of these larger sizes should not be ordered for, or held in stock.

Orders must be placed some time in advance, as special arrangements have to be made to manufacture the larger sizes.

SOCKETS.—All the tubes except those of 20 gauge may be used as sockets for other standard tubes.

STRUTS.—The lower curve (Fig. 128) gives the limiting stress, and hence the limiting load, for a tubular strut of any size and length, allowing for the crookedness and eccentricity of bore allowed in the Air Board Tube Specifications.

SPECIALLY STRAIGHT TUBES.—The tubes in this schedule can be supplied, when so ordered, with a deviation from straightness not exceeding one-half the amount allowed in the Tube Specifications (which is the same as that given in *Recent Researches on Tubular Struts*). The upper curve on Fig. 128 gives the limiting stress, and hence the limiting load, for struts of any size and length allowing for this reduced crookedness and the eccentricity of bore allowed in the Air Board Specifications.

TABLE 93

SUPPLEMENTARY SCHEDULE OF STANDARD SIZES OF STEEL TUBES FOR AIRCRAFT

TUBES FOR STRUTS, STRAIGHTENED AND NORMALISED

Outside Diameter (Tolerance ± ·004 in.)	To Take.	Minimum Area of Section (see Note).	Maximum Weight per Foot (see Note).	Radius of Gyration (see Note).	Outside Diameter (Tolerance ± ·004 in.)	To Take.	Minimum Area of Section (see Note).	Maximum Weight per Foot (see Note).	Radius of Gyration (see Note).
Ins.	Ins.	Sq. Ins.	Lbs.	Ins.	Ins.	Ins.	Sq. Ins.	Lbs.	Ins.
		Approximately 11 Gauge, Thickness ·118 in. Min. to ·122 in. Max.							
2¼	2	·790	2·719	·754	3¾	3½	1·347	4·735	1·283
2½	2¼	·884	3·105	·841	4	3¾	1·439	5·061	1·371
2¾	2½	·976	3·431	·930	4¼	4	1·532	5·387	1·460
3	2¾	1·068	3·757	1·018	4½	4¼	1·625	5·713	1·548
3¼	3	1·161	4·083	1·106	4¾	4½	1·717	6·039	1·636
3½	3¼	1·254	4·409	1·195	5	4¾	1·810	6·365	1·725
		Approximately 17 Gauge, Thickness ·056 in. Min. to ·060 in. Max.							
2¼	2⅛	·386	1·407	·774	3¾	3⅝	·650	2·369	1·304
2⅜	2¼	·408	1·487	·818	3⅞	3¾	·672	2·449	1·348
2½	2⅜	·430	1·567	·862	4	3⅞	·694	2·529	1·393
2⅝	2½	·452	1·647	·907	4⅛	4	·716	2·609	1·437
2¾	2⅝	·474	1·727	·951	4¼	4⅛	·738	2·689	1·481
2⅞	2¾	·496	1·808	·995	4⅜	4¼	·760	2·769	1·525
3	2⅞	·518	1·888	1·040	4½	4⅜	·782	2·850	1·569
3⅛	3	·540	1·968	1·084	4⅝	4½	·804	2·930	1·614
3¼	3⅛	·562	2·048	1·128	4¾	4⅝	·826	3·010	1·658
3⅜	3¼	·584	2·128	1·172	4⅞	4¾	·848	3·090	1·702
3½	3⅜	·606	2·208	1·216	5	4⅞	·870	3·170	1·746
3⅝	3½	·628	2·288	1·260					
		Approximately 20 Gauge, Thickness about 1 mm., from ·036 in. Min. to ·040 in. Max.							
2¼		·250	·946	·781	3		·335	1·267	1·046
2½		·278	1·053	·869	3¼		·363	1·374	1·135
2¾		·307	1·160	·958	3½		·392	1·481	1·223
		Approximately 22 Gauge, Thickness ·025 in. Min. to ·029 in. Max.							
2¼	2³⁄₁₆	·174	·689	·785	2⅝	2⁹⁄₁₆	·204	·806	·917
2⅜	2⁵⁄₁₆	·184	·728	·829	2¾	2¹¹⁄₁₆	·214	·844	·962
2½	2⁷⁄₁₆	·194	·767	·873	2⅞	2¹³⁄₁₆	·224	·883	1·006
					3	2¹⁵⁄₁₆	·233	·922	1·050

EXTRA TUBES FOR SOCKETS ONLY

Approximately 22 Gauge, Thickness ·025 in. Min. to ·029 in. Max.

Outside Diameter (Tolerance ± ·004 in.)	To Take.	Minimum Area of Section (see Note).	Maximum Weight per Foot (see Note).	Radius of Gyration (see Note).	Outside Diameter (Tolerance ± ·004 in.)	To Take.	Minimum Area of Section (see Note).	Maximum Weight per Foot (see Note).	Radius of Gyration (see Note).
2¹⁄₁₆	2	·160	·670		2¹¹⁄₁₆	2⅝	·209	·825	
2³⁄₁₆	2⅛	·170	·709		2¹³⁄₁₆	2¾	·219	·864	
2⁵⁄₁₆	2¼	·180	·747		2¹⁵⁄₁₆	2⅞	·228	·902	
2⁷⁄₁₆	2⅜	·190	·786		3¹⁄₁₆	3	·238	·941	

NOTE.—Sectional areas are calculated for tubes of minimum thickness, viz., ·118, ·056, ·036, and ·025 in. respectively.

Weights per foot are calculated for maximum thickness, viz., ·122, ·060, ·040, and ·029 in. respectively.

Radii of gyration are calculated for minimum diameter and thickness.

LIMITING STRESS FOR TUBULAR STEEL STRUTS WITH ROUND OR PIN-JOINTED ENDS

Yield point of steel . 28 tons per square inch
Young's Modulus, E . 13600 " " "

(*A.D. Specification T*.1.)

L = Length of strut in inches.
K = Radius of gyration of cross section (*see* Table 36).
Limiting load = limiting stress × area of cross section (Table 36).
Curve A for tubes of ordinary crookedness (to A.D. Specifications).
Curve B for specially straight tubes.
Curve E is Euler's curve.

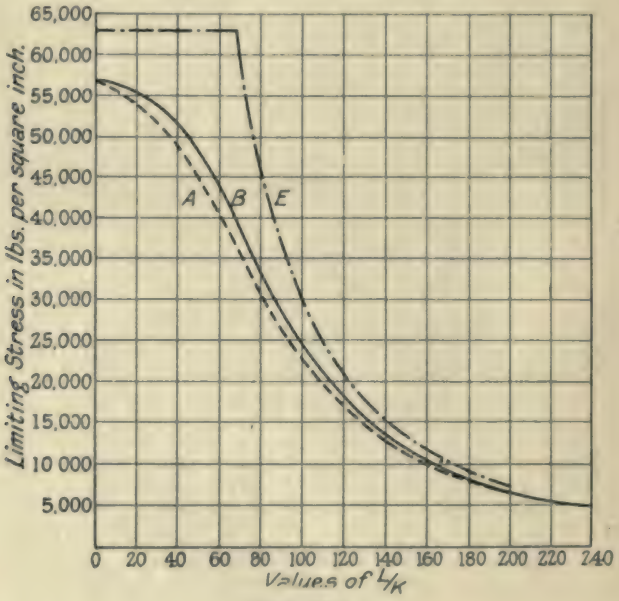


FIG. 128

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